# Final Design for a Comprehensive Orbital Debris Management Program

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## Executive Summary

STRES, Incorporated has recently been contracted for the design of a comprehensive program for the control of orbital debris. This document describes the rationale and specifics of this type of design, as well as details the various components of the overall plan.

The problem of orbital debris has been steadily worsening since the first successful launch in 1957. Currently countries spend billions of dollars in attempts to shield space operations from damage caused by debris, and still these operations are often rendered useless by damage from collisions with debris. The hazards posed by orbital debris suggest the need for a progressive plan for the prevention of future debris, as well as the reduction of the current debris level. The proposed debris management plan includes debris removal systems and preventative techniques and policies.

The debris removal is directed at improving the current debris environment. Because of the variance in sizes of debris a single system cannot reasonably remove all kinds of debris. An active removal system, which deliberately retrieves targeted debris from known orbits, was determined to be effective in the disposal of debris tracked directly from earth. However, no effective system is currently available to remove the untrackable debris.

The debris prevention program is intended to protect the orbital environment from future abuses. This portion of the plan involves various methods and rules for future prevention of debris. The preventative techniques are protective methods that can be used in future design of payloads. The prevention policies are rules which should be employed to force the prevention of orbital debris.

The design process was governed by a management structure headed by a project manager and a technical manager. The project was completed on time and $7,432 over budget.

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## List of Acronyms

| | |
|---|---|
| ASAT | Anti-satellite |
| DPS | Data Processing System |
| DUS | Deployable Umbrella Satellite |
| GNC | Guidance, Navigation and Control |
| IMU | Inertial Measurement Unit |
| LADAR | Laser Detection and Ranging |
| LBU | Laser Beam Unit |
| LDEF | Long Duration Exposure Facility |
| LEO | Low Earth Orbit |
| NASA | National Aeronautics and Space Administration |
| NORAD | North American Air Defense Command |
| OMV | Orbital Maneuvering Vehicle |
| POV | Proximity Operations Vehicle |
| RCS | Reaction Control System |
| RMS | Remote Manipulator System |
| ROBS | Resuppliable Orbital Base System |
| RTG | Radioisotope Thermoelectric Generator |
| RRS | Resuppliable Roving Systems |
| RS | Repulsor Satellite |
| RV | Roving Vehicle |
| Solar MAX | Solar Maximum Mission Satellite |
| SRRV | Single Rendezvous and Return Vehicle |
| STRES | Space Trash Removal and Elimination Systems |
| USSPACECOM | United States Space Command |
| UN | United Nations |
| UNCOPUOS | United Nations Committee on the Peaceful Uses of Outer Space |

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## 1.0 INTRODUCTION

Due to the increasing hazards of orbital debris, Space Trash Removal and Elimination Systems (STRES), Incorporated was contracted under proposal #ASE274L to submit a comprehensive program for the control of orbital debris. STRES, Inc. will provide a plan for removal and prevention of debris as a method of reducing the danger to space personnel and active operations.

The proposed orbital debris management program will provide for preventive measures for future space operations as well as provide a comprehensive scheme for removal of current and future debris. Unfortunately, systems which can remove all orbital debris are currently unrealistic; therefore, such performance from a system will not be required. However, a removal system must reduce debris significantly in order to be effective. Furthermore, the technology to prevent the creation of all future debris is currently unavailable; accordingly, prevention techniques will be expected to significantly reduce any future contributions to debris.

In order to promote success of an orbital debris removal program, the concept of debris management must have public support. Three basic factors which affect support of space ventures are initial and overall cost, measurable benefits, and environmental safety. Since orbital debris removal does not have sufficient economic benefit to be supported by private industry, the primary source of funding will be federal treasuries. Therefore, it is essential to minimize the cost of the system to receive support. In accordance with the funding difficulties, it is necessary to show direct and measurable benefits to justify the expense of orbital debris management. Visible benefits, such as protecting the proposed space station *Freedom*, would foster public support, and thereby encourage funding. Finally, the environmental safety of any endeavor is paramount. Due to the public's growing awareness and concern about environmental hazards and risks, all management techniques must be safe for the earth's atmosphere and orbital environment to maintain public support.

## 2.0 ORBITAL DEBRIS BACKGROUND AND SEVERITY

In order to design an appropriate management system for orbital debris, it is necessary to understand the historical background as well as the current severity of the problem.

### 2.1 Background

Scientists have been concerned about the dangers posed by orbital debris since the Apollo and Gemini missions. During these early ventures, the main concern was the possible hazards of natural orbital debris, such as meteoroids and cosmic dust. However, after an in-depth study, it was concluded that spacecraft could be effectively and efficiently shielded from the dangers of natural debris because of its small size and density. Since that time, concern has shifted to the dangers of manmade, rather than natural, debris. The main reason for the change in focus is that the level of manmade debris has far surpassed the level of natural debris in the orbital environment. In addition, most manmade debris is significantly larger in size than the average meteoroid, and therefore, is much more dangerous to the spacecraft. Finally, there are few meteoroids in earth orbit because they generally only pass through the earth's influence, whereas manmade orbital debris remains in earth orbit until its orbit decays and the debris re-enters the earth's atmosphere. Namely, the flux of natural debris is significantly less than the flux of manmade debris as illustrated in Figure 1.

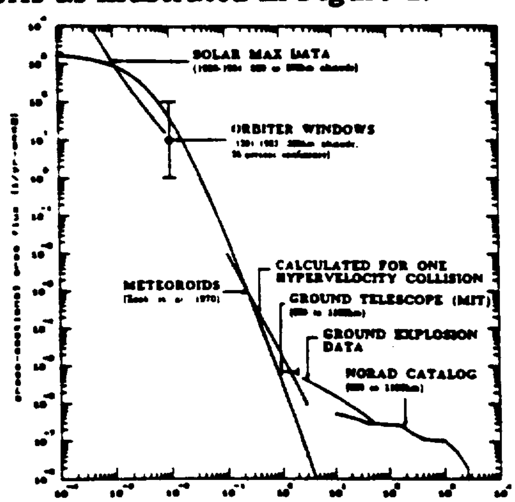


**Figure 1. Natural Debris Flux Versus Manmade Debris Flux [12:10]**



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One of the early misconceptions about orbital debris was that anything placed into earth orbit would eventually return to earth. However, it requires over 1000 years for a payload at an altitude of 1000 km to deorbit, and an object in a 10,000 km orbit may never deorbit.[15:176] Since the natural process of removing orbital debris is so slow, and because there is currently no orbital debris removal system available, the amount of orbital debris is steadily increasing. If action is not taken to reduce the contributions to debris levels, the use of space may be endangered within a decade or two and even several lower earth orbits may be rendered completely unusable.

### 2.1.1 Types of Space Debris

Another common misconception about the orbital debris problem is that there is only one type of orbital debris. Actually, there are several different types of manmade debris that are the result of unique sources.

#### 2.1.1.1 Mission Related Debris

Since the launch of Sputnik I, almost 7500 mission-related objects have been deposited in space. Mission related debris includes hardware, protective equipment, and even waste.

A main contributor the the orbital debris problem is the protective equipment, such as paint and shields, installed on spacecraft. Frequently, paint is degraded by the orbital environment. Once it begins to flake off of the spacecraft, it becomes a hazard to other spacecraft as well as to itself. Another contributor is the shielding used to protect most satellites from orbital debris. These shields fragment when impacted by debris as small as 1 mm in diameter. The fragments from the shield can often pose a greater hazard to the satellite and other spacecraft than that posed by the debris with which it impacted.

Another cause of debris is mission related hardware. This category of debris consists mostly of staging mechanisms, such as explosive bolts and separation rings, as well as protective shields which are shed during deployment of a payload. Even the emissions from solid rockets contribute significantly to the debris population. However, a majority of the mass is a result of spent upper stage rockets left in orbit by past missions.

Finally, litter from various manned missions has added to the amount of orbital debris. For example, U.S.S.R. cosmonauts often jettisoned bags of garbage containing dirty cloths, food wrappers, and other trash from the Salyut 7 space station. Even crystalized urine from the shuttle was discovered on the Solar Max.[21:3.50]

#### 2.1.1.2 Explosion Remnant Debris

As of February 1988, nearly 90 intentional or accidental catalogued payload explosions had deposited more than 36,000 kg of debris fragments into space, with a significant portion of this mass in the 1 mm to 10 cm untrackable range. These fragmentations account for nearly 40 percent of all tracked objects greater than 10 cm in diameter,[9] and have contributed an estimated 30,000 to 70,000 fragments in the 1 to 10 cm range of debris.[6]

Deliberate explosions for military and intelligence operations are a primary cause of fragmentation debris. Satellites have been deliberately destroyed in orbit to prevent the recovery of certain payloads and to test military hardware. For example, the U.S.S.R. intentionally exploded Kosmos 1813 on January 29, 1987 to prevent possible recovery by the United States.[8:51] Additionally, the anti-satellite (ASAT) programs of the United States and the U.S.S.R. are responsible for the deliberate destruction of satellites and the creation of orbital debris. Every ASAT test is capable of producing up to 10 million particles. Since the beginning of the ASAT programs, there have been sixteen tests conducted by the U.S.S.R. alone.

Besides the intentional destruction of satellites, there is also the possibility of accidental satellite explosions. Inadvertent fragmentation is generally the result of propulsion system failures. The best example of such a failure is the second stages of the United States' Delta rockets. Since 1973, seven Delta second stages have exploded and produced 1230 known orbital debris objects after successfully performing their payload delivery missions.[11:17] More importantly, some of these rockets were presumed dead for as long as three years prior to exploding. After thorough investigation, it was revealed that residual hypergolic propellant was responsible for these detonations. Other examples are the explosions of the Ariane third stages that occurred in November 1986 and February 1990. The 1986 explosion is considered the worst breakup in history. Launched in February 1986, the booster exploded at an altitude of 780 km, and created more than 200 pieces of large trackable debris in orbits ranging from 430 to 1430 km altitude.[25:34] Even though the cause of this explosion is still undetermined, possible causes and results of explosions are illustrated in Figure 2.

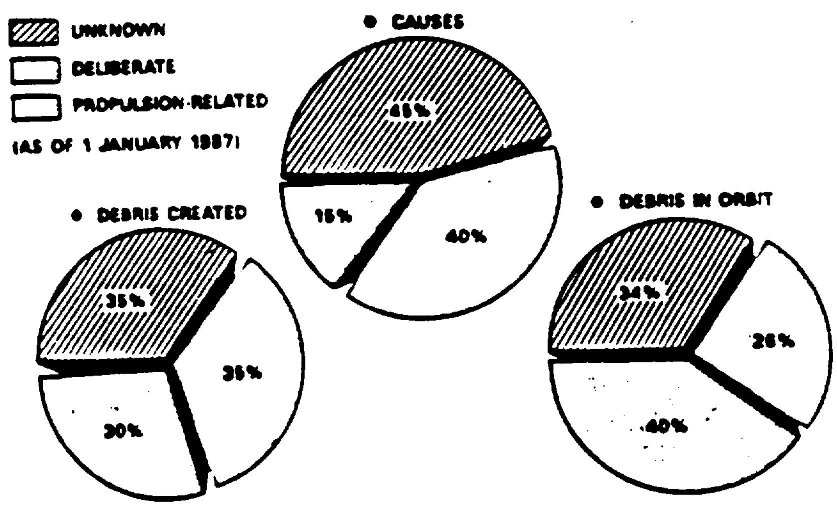


**Figure 2. Causes of Satellite Fragmentations [5:20]**

### 2.1.2 Evidence of Damage from Space Debris

The first indication that orbital debris was colliding with active payloads was obtained from Explorer 46. Launched in August 1972, this satellite included a meteoroid bumper experiment which was sensitive to impacts by particles larger than 0.1 mm. Data from this experiment suggested that 43 of the impacts experienced by Explorer could have been the result of manmade debris because the corresponding orbital debris flux experienced was three times greater than that expected for natural debris.[14:22]

The first conclusive proof that orbital debris was striking active payloads was provided by the Skylab cosmic dust experiment. The sole purpose for this experiment was to analyze meteoroid impacts. Chemical analysis revealed high levels of aluminum in the impact craters. Additionally, windows on the returned Skylab IV Apollo Module were examined for meteoroid impacts. It was discovered that about 50% of the hypervelocity pits covering the windows were aluminum lined, probably the result of collisions with aluminum oxide particles, which are the primary constituents of the thermal coatings used on most spacecraft.[14:23]

The best known example of damage due to orbital debris is the damage to Space Shuttle Challenger during Space Shuttle Mission 31-C. Space Shuttle Challenger's forward window was impacted by a small piece of debris that left a crater 2.0 mm across and 0.63 mm deep in the window. This impact is believed to be the first confirmed damage to an operational space vehicle by orbital debris. At first believed to be a micrometeorite impact, it was later determined that the object was a piece of thermal paint about 0.2 mm in diameter that struck the glass at a speed of 4-6 km/s.[10:89] Due to the severity of the impact, the window had to be replaced at a cost of more than $50,000.

Most recently, examination of insulation louvers recovered from the Solar Maximum Mission Satellite (Solar MAX) has revealed extensive hypervelocity impacts with meteoritic material, paint particles, solid rocket emissions, and particles of unknown origin. Studies indicate that at least 70% of these impacts were caused by manmade orbital debris.

## 2.2 Current Problem

Currently, the North American Air Defense Command (NORAD) tracks over 7000 objects ranging in altitude from 100 km to 100,000 km. Of these objects, only 5% are operational payloads, while the remainder constitute an orbiting junkyard of inactive satellites, discarded equipment, and large fragments from payload breakups. As of April 1, 1989, inactive payloads accounted for 21% of the trackable population, rocket bodies and launch debris, 31%, and fragmentation debris 43%.[15:17] The various types of trackable objects can be seen in Figure 3.

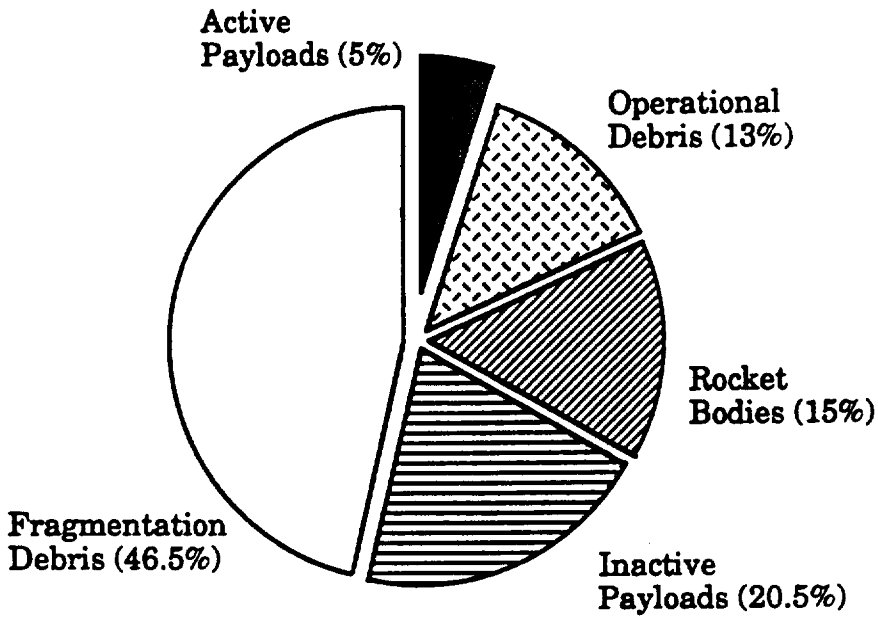


**Figure 3. Types of Trackable Objects [15:17]**

Furthermore, the number of trackable objects has been increasing at a rate of nearly 300 per year, despite an average international launch rate of only 121 launches per year.[15:16] The fact that the trackable orbital population is increasing faster than the average launch rate is attributed to launch equipment and satellite explosions. The trend of trackable objects in space is shown in Figure 4.

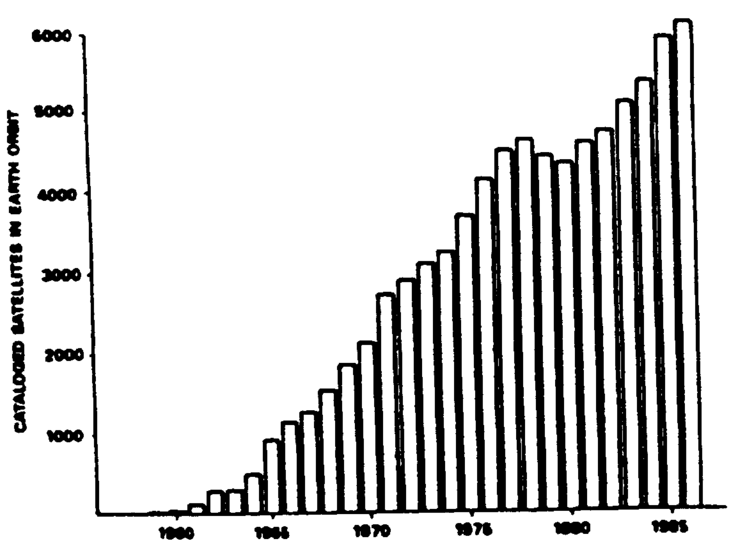


**Figure 4. Growth Rate of Catalogued Objects [15:16]**

In addition to these trackable items, NORAD estimates that there are billions, possibly even trillions, of pieces of microparticulate matter varying in size from 1-100 microns.[26:3] This type of debris is created from various sources, including solid-propellant rocket motors and spacecraft coating degradation due to ultraviolet radiation and atomic oxygen.

At the current time, there are several satellite breakups for officially unknown reasons that are suspected to be the result of collision with orbital debris. Specifically, the failure of the U.S.S.R. Kosmos 954 in 1978 was attributed to orbital debris by Soviet officials [4:24], and U.S.S.R. Kosmos 1275 may have been completely destroyed by a collision with orbital debris [16]. Each of these breakups resulted in the creation of at least 1000 objects that were approximately 10 cm in size. Each of these secondary fragments is capable of causing a catastrophic failure of another payload. Concern about the overabundance of these objects has given rise to fears that orbital debris will become self-generative. From these examples, it appears that debris has already achieved that level in some lower earth orbits. Therefore, if the debris problem is not addressed, it could soon become unmanageable and detrimental to future payloads.

### 2.2.1 Location of Space Debris

In order to develop the most efficient measures for combatting the debris problem, it is necessary to know the locations and densities of debris.

The greatest concentration of orbital debris is found in low earth orbit (LEO). LEO is defined as a spherical shell, bounded by altitudes of 200 km and 4,000 km. As of January 1988, 83% of the approximately 7,000 tracked objects resided in orbits with an average altitude below 6,000 km; however, the specific density of debris varies with altitude and inclination.[11:17] The spatial density of orbital debris, shown in Figure 5, is highest at inclinations of 32°, 66°, 74°, 82°, 91°, and 100° due to numerous launches and due to several large scale breakups in these regions.

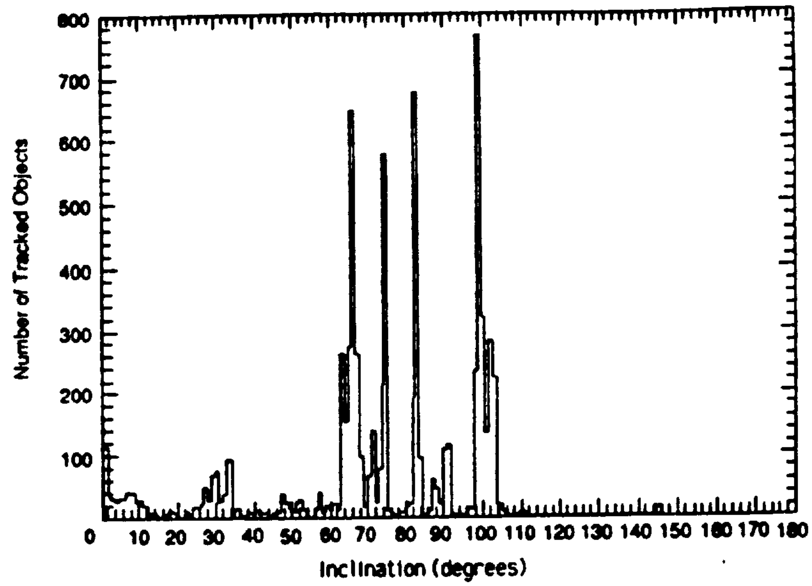


**Figure 5. Spatial Density of Orbital Debris at Various Inclinations [21:1.9]**

The distributions of large and small debris are generally the same for the various inclinations; however, they do vary with altitude. The concentration of large orbital debris, shown in Figure 6, is between altitudes of 175 km and 1000 km; however, since objects below 400 km quickly enter due to atmospheric drag, only altitudes above this are of concern. Once outside this range, the density of debris drops dramatically. In contrast, the density of small debris, shown in Figure 7, reaches a maximum around the 500 km orbits, and tends to be more uniformly distributed according to altitude than the large debris.

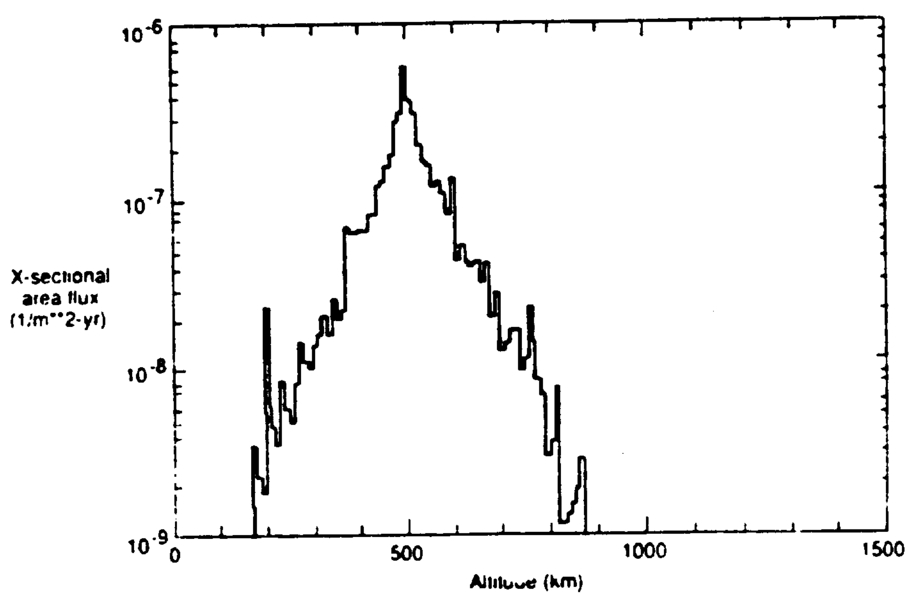


**Figure 6. Flux Variation for Large Debris at Various Altitudes [21:4.24]**

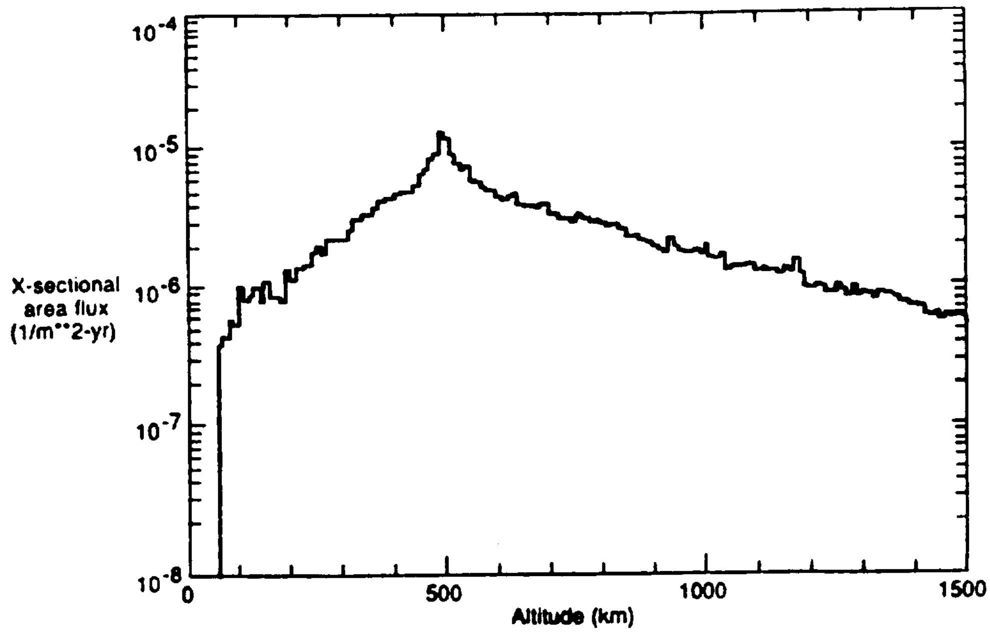


**Figure 7. Flux Variation for Small Debris at Various Altitudes [21:4.25]**

There has been concern expressed about the levels of orbital debris at the geosynchronous orbits. Currently this concern is unjustified since the hazard from orbital debris in these orbits is less than the hazards posed by natural meteoroids passing through the orbit.[21:16] The generally accepted standard is that any danger less than that posed by natural sources is insignificant and should not be of concern.

2.2.2 Hazards of Space Debris

In the past 10 years, the risk of a collision has increased by four orders of magnitude.[22:F4] The risks posed by orbital debris can be broken down into two major categories. The first category is the danger of collision with primary debris. This type of debris includes inoperative satellites, spent rocket motors, or microscopic particles directly from these primary objects. The second category is the possibility of collision with secondary debris. Secondary debris results from collisions between two primary objects. The problems from secondary debris lead to the possibility of the cascade effect.[12:2637] This phenomenon is characterized by the exponential growth of orbital debris caused by uncontrolled collisions between debris.

### 2.2.2.1 Primary Debris

The major risks that orbital debris poses to space personnel and active payloads are dictated by the probability of collision or degradation. A collision may result in the loss of property or life, generation of more debris, release of contamination, or the failure of mechanical parts. Deterioration of components of orbital operations and space activities will only increase as the amount of orbital debris increases.

The physical danger from orbital debris varies with the size and velocity of debris objects encountered. As illustrated in Table 1, possible damage can range from the loss of subsystem capabilities to spacecraft obliteration. Degradation of spacecraft capability may occur due to pitting or fracturing of protective surfaces such as solar cell cover glasses or special thermal coatings. In addition, the skin of a spacecraft could be penetrated, leading to damage or destruction of subsystem components or even high-pressure fuel tanks and propulsion systems.

**Table 1. Summary of Risks From Debris [10]**

| Debris Size (mass) | Nature of Threat | Relative Probability |
|---|---|---|
| Submillimeter (microgram) | Degrade Optics, Solar Panels | Most Probable |
| Millimeter (milligram) | Penetrate unshielded satellite or space craft | Less Probable than above |
| Centimeter (gram) | Penetrate shielded satellite or spacecraft | Less Probable than above |
| Decimeter (kilogram) | Fragment satellite or spacecraft | Least Probable |

Aside from the risk of mechanical destruction, the possibility of injury to personnel is a serious danger. With velocities averaging 10 km/s, even small debris particles could seriously injure or even kill an astronaut. Hypervelocity tests show that a 0.5-mm paint chip traveling at 10 km/s could easily penetrate a standard spacesuit and kill an astronaut engaged in extravehicular activity.[23:187] Astronauts are not even sufficiently protected in a vehicle. For example, if a debris object 1 cm in diameter traveling 10 km/s struck a space station, it could penetrate a pressurized crew module, decompress the module, kill the crew, and could eventually lead to the breakup of the station.

#### 2.2.2.2 Secondary Debris

Perhaps the most serious consequence of collision with orbital debris is the generation of secondary debris. This phenomenon known as the cascade or Kessler Effect, was first hypothesized by Donald J. Kessler of the National Aeronautics and Space Administration (NASA) in 1978. Kessler, Project Scientist for Orbital Debris at NASA, theorized that as the number of space objects in earth orbit increases, the probability of collisions between them also increases. Moreover, collisions between these primary objects could produce new secondary fragments. When sufficient secondary debris has been generated, the debris flux will increase exponentially with time even if no new objects are placed into orbit.

Currently, experts predict that the levels of debris need only be two to three times the current levels to cause exponential space debris growth. Additionally, if the current trends continue, these levels of debris could be reached in twenty to fifty years.[6:1] The end result would be the formation of a debris belt around the earth that could seriously impair utilization of space.

### 2.2.3 Probability of Collision with Space Debris

Calculating the probability of a collision with orbital debris is important, not only for safety considerations, but also for determining the economic and political costs for future space activities. Establishing the likelihood that a particular event will occur and the extent of the resulting damage, has not yet been perfected; however, it does provide a good measure for mission planning purposes. One of the primary reasons for the uncertainty in prediction is that the spatial density of all debris is not known due to NORAD's limited detection capability as illustrated in Figure 8.

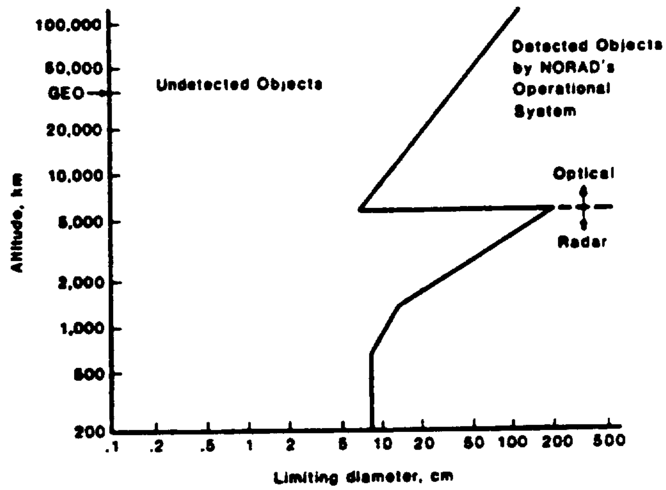


**Figure 8. NORAD's Operational Detection Ability [15:7]**

Recent research strongly indicates that previous calculations of the probability of collisions in LEO between orbital debris and active payloads were conservative. Although there are currently 7100 trackable objects, the total number of objects in LEO has been calculated to be between eight and eleven times higher in order to compensate for the limitations on tracking capability.[20:C1] Recent observations, such as those made from Solar MAX, indicate that the debris population may be even higher.

The size of a payload is a principal factor in determining its chance for a collision. Therefore, the collision risk is lowest for the small communication and unmanned satellites. Although collisions between orbital debris and these active satellites were not considered to be significant as late as 1984, debris particles traveling between 7-10 km/s are now believed to pose a significant danger to such active satellites. The probability of a catastrophic collision between a functioning satellite and orbital debris by 1995 in the most densely populated region of LEO (900-1000 km) is currently estimated to be 63%.[7:47]

For the larger space objects, the probability of collision increases. For example, it is currently estimated that a space station with a cross-sectional area of 1 square km orbiting at 500 km in an inclination of 28.5°, will be hit by orbital debris at least once a year. If the station is placed at 1,000 km, the probability of impact increases to 20 times per year.[17:41] Since the extent of damage will depend on where the orbital debris strikes and the size of the impacting object, these numbers show that space station *Freedom* has a significant chance of being impacted and seriously damaged by orbital debris.

The risk of collision grows exponentially with the growth rate of debris, as shown in Figure 9. In 1982, NASA calculated that orbital debris was increasing at about 13% per year.[8:47] At this rate, it is estimated that the orbital debris population would double in the next 10 years and would increase the collision hazard eight-fold in 20 years. In 1986 it was determined that if past growth rates continued, collisions between objects larger than 4 cm could be expected within a few years.

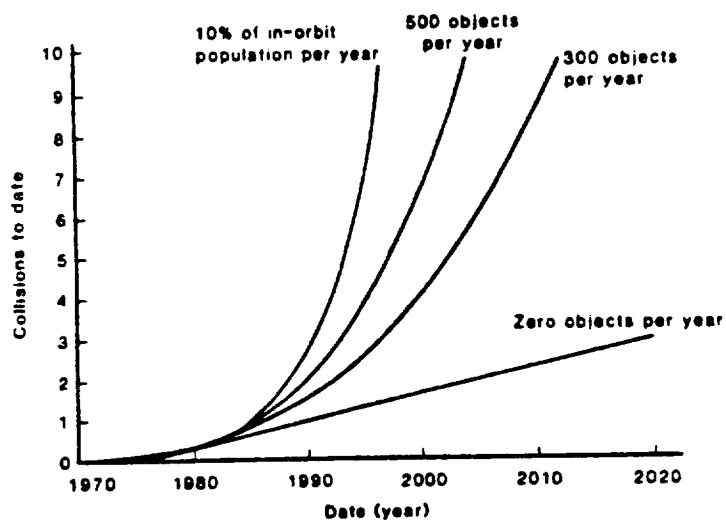


**Figure 9. Collisions Based on Estimated Debris Growth Rates [15:7]**

### 2.2.4 Mission Modifications due to Space Debris

Risks of collisions with orbital debris is becoming a significant factor when designing spacecraft and spacesuits. The potential hazard to humans and active payloads has resulted in the alteration of operations and design.

Space debris also forces modifications when planning the trajectory for current and future spacecraft. Before a United States owned satellite is launched, the orbit of every catalogued space object must be examined to ensure the spacecraft will not pass near any orbital debris during its first few hours in orbit. Although collision avoidance for payloads throughout their active lives is impossible, satellite management programs will need to be modified if the quantity of orbital debris increases as predicted. The United States Space Command (USSPACECOM) and NASA are currently working together to determine the feasibility of maneuvering the proposed space station *Freedom* to avoid collisions. The possibility that a spacecraft will have to consume fuel to avoid orbital debris increases the required propellant and, as a result, increases the overall mission cost.

### 2.3 Expected Debris Growth Rate

In the past thirty years, the growth rate of catalogued objects has been relatively constant, as shown in Figure 4. In order to establish a method of managing this growth, it is essential to determine controlling variables. According to Kessler [21:4.33], the growth rate of the debris population depends on various sources and sinks. This relationship can be modeled by a quadratic equation, where

$$\begin{aligned} dN/dt &= \text{Sources} - \text{Sinks} \\ &= A + BN + CN^2 \qquad [1] \end{aligned}$$

where

N = debris population (in arbitrary units)
A = most space operations
= deployment + launch - retrieval - deorbits
B = processes related to number of objects
= explosions - debris sweeper
C = interaction process(collisions).

According to this model, the growth rate, as shown in Figure 10, is stable until the debris population reaches $N_2$, at which time it becomes unbounded. Therefore, it is essential to increase $N_2$.

$0 \le N \le N_1$: Growth to $N_1$, then Stable.

$0 \le N \le N_2$: Decline to $N_1$, Stable.

$N \ge N_2$: Runaway Growth.

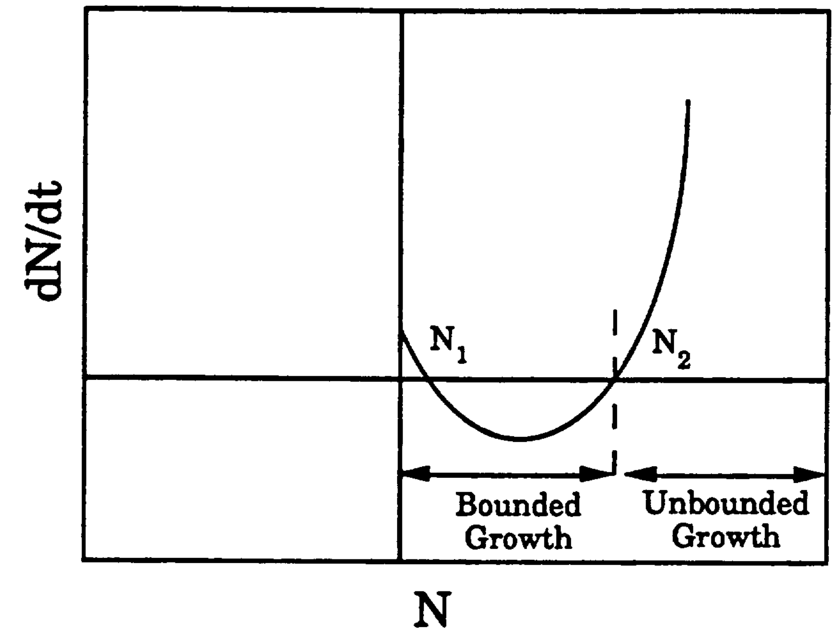


**Figure 10. Kessler's Model of the Debris Population Growth [21:4.34]**

Since the quantity of space operations, A, determines the position of the growth curve, A is the main determining factor in the values of $N_1$ and $N_2$. Hence the removal of large objects at the end of their useful lives may determine whether the debris population is controlled or self-generating. Although the number of launches and retrievals is the main factor, the reduction of explosions and small debris is also effective in determining $N_1$ and $N_2$, as well as it has a significant affect on C which cannot controlled.

From this model, it is apparent how the activities in space can significantly affect the debris population. This demonstrates the need for an effective and progressive debris management program.

## 3.0 SYSTEM SELECTION

When choosing the preferable system, the overall cost is the deciding factor. The overall cost of a system is determined by direct and indirect costs, where the direct costs include the mission equipment, maintenance, and repair, while the indirect costs include research, design, and environmental degradation.

### 3.1 Direct Costs

The direct costs of a space venture involve any technological developments or equipment required for success. The development costs include research, design, and testing, while equipment costs are determined by the system weight and size, the number and type of launch vehicle, and the expected lifetime.

#### 3.1.1 Technological Developments

The cost of developing new technology and proving its capabilities and reliability often far exceeds the actual equipment costs. Due to the exorbitant costs of technological developments, it is not prudent to propose a design which is based on unavailable technology.

In order to foster public support, a launch date of 1996 has been selected with the rationale of beginning clean up of the orbital environment before construction begins on space station *Freedom*. Because of the targeted launch date of 1996, it is important that the technology used for a solution be currently available. In addition testing costs can be reduced by selecting technologies that have been verified and flight proven.

#### 3.1.2 System Weight and Size

Because of the difficulties and limitations of launching and maintaining massive space structures, the cost of equipment increases greatly with slight increases in weight and size. Therefore any reduction in either weight or size is an important achievement. The cost can quickly be reduced by minimizing the weight because it costs between $6000 and $10,000 per pound to place a system in orbit. Hence, reducing the size of a system also reduces costs because of the cost and difficulty of deployment of large systems and the fuel required for station keeping to offset the atmospheric and solar drag on large systems.

### 3.1.3 Launch Vehicle

Because of the various capabilities and costs of launch vehicles, the selection is an important decision. Since the cost of launching is enormous, and the price of a launch vehicle generally increases with its capacity, an efficient use of a launch vehicle payload capacity is paramount. Specifically, a special focus has been placed on use of the 450 decommissioned Minuteman II missiles. These vehicles are to be decommissioned with the next few years; therefore they may be a viable and inexpensive solution to the launch vehicle problem. Finally, decreasing the total number of launches necessary for the system to achieve a given level of debris reduction will further reduce the cost because of the high cost of launch vehicles.

### 3.1.4 Lifetime

The direct cost for a system is easily appraised, but often it is sorely underestimated. Included in the reasons for this are the tendency to ignore the cost of replacement and repair of equipment. Maximizing the expected lifetime of a system and its components significantly decrease the cost of replacement and repair. Therefore it is essential to choose a system and subsystems which are reliable and long lasting.

## 3.2 Indirect Costs

The direct costs of a mission are the most visible, however they may not be the highest. Often, the detrimental effects on the environment cannot be measured with simple monetary figures. Although the monetary costs are obvious concerns, the environmental costs of a system are becoming more visible. These environmental costs include adverse effects on both the earth's atmosphere and orbital environment. This means a system should not pollute, degrade, or endanger either setting. Specifics which must be accounted for are the possibilities of damage to earth surface property, the danger to active orbital operations, the contribution to orbital debris, the degradation of the ozone layer, and the contamination of the atmosphere.

### 3.2.1 Property Damage

A main concern about the removal of debris is the potential hazards to property. Damage may occur to either the earth's surface or to orbital property, and it has the potential of being extremely expensive, or even fatal.

First, controlling entry of all large debris reduces the possibility of damage to earth surface property or active operations. While on a deorbit trajectory, if debris is not guided, it may hamper or even destroy active orbital operations. In addition, if debris is very large, there exists the possibility that it will not completely burn up during entry, and if the entry is not controlled, or improperly controlled, there exists the possibility that the debris will cause damage to property, rather than falling safely into an ocean.

For example, there are at least three incidents of earth based property damage. The first occurred in 1969, when a Japanese ship was struck by falling debris from a U.S.S.R. rocket, and five people were injured. Secondly, the U.S.S.R. Kosmos 954 equipped with a nuclear power source landed in sparsely populated areas in Canada in 1978. This incident caused great concern about the environmental effect as well as direct damage to property or persons.[1:403] Finally, the United States Skylab entered over Australia, and the largest piece to land was over 1000 pounds. [6]

Aside from the hazards of randomly falling debris, there is the issue of restitution for any damage. Because a country is responsible for damage from active space operations and since the debris deorbit is intentional and active, the supporting country will be held liable for all damage.

### 3.2.2 Orbital Debris

Another measure of a system's effect on the environment is the amount of additional orbital debris it produces. Since the objective of a system is the reduction of debris, if a system contributes to the debris problem then it is defeating the purpose. Consequently, it is imperative that a system is clean.

Generally, launch vehicles employ solid rocket boosters because they have a much greater thrust over a short span of time than do liquid fuel engines. However, solid rocket motors eject large pieces of unburned fuel as well as billions of particles of aluminum oxide creating clouds of debris that linger and present substantial hazards to orbital operations.[2:80] Because these solid motors increase the amount of particulate debris, the number of launches should be minimized in order to maintain the effectiveness of a system.

### 3.2.3 Ozone Depletion

The environmental impact of launching large numbers of solid rocket boosters is a growing concern in the global community. The original shuttle schedule, which predicted approximately thirty launches per year, drew fire from environmentalists and scientists alike for its reliance on solid rocket boosters during launch. Similarly, the use of the Minuteman II, one of the dirtiest solid rocket boosters, will place a debris management plan under scrutiny.

Launch vehicles generally employ solid rocket boosters because of their lower cost and greater thrust capability. For solid propellents, the oxidizer is the major portion of the composition, yet it is this oxidizer that is the most harmful to man and produces by-products detrimental to the ozone layer. This hazard has drawn attention amid recent discoveries of declining global ozone levels. In fact, *Space News* reported the findings by two Soviet scientists that suggested a catastrophic loss of the entire ozone if 300 shuttle missions were carried out annually [5:8]. They also predicted that if the use of solid rockets continues at the current rate, ozone levels will decrease by 10% over the next 15 years. Even though the chloroflourocarbons, and other ozone depleting substances, released each year by the United States electronics industry are far greater than those produced by reasonable launch activity, a debris management program must minimize launches in order to avoid public disdain.

### 3.2.4 Nuclear Contamination

Finally the possibility of atmospheric contamination must be addressed. Since nuclear satellites may be candidates for removal, the safety of vaporizing nuclear matter over the global atmosphere must be studied.

This impending danger from nuclear satellites, has been another major concern of scientists over the last several decades. Since 1961, the United States has launched one nuclear reactor and 38 radioisotope thermoelectric generators (RTGs) as power sources. Of these, the nuclear reactor and 10 of the RTGs remain in earth orbit. Additionally, the U.S.S.R has 29 nuclear reactors and over 1400 kg of nuclear material (mostly spent cores) still in orbit [3:91]. NUS Corporation has estimated that 3 previous vaporizations of nuclear payloads over populous areas will cause an added 6.72 cancer deaths over the next 20 to 30 years.[4:153] However, these deaths could be all but eliminated by properly deorbiting nuclear matter to remote areas of the earth .

Losses due to nuclear vaporization are minute compared to the estimated 900,000 to 1,400,000 deaths in the same time period due to natural radiation and nuclear testing fallout.[7:156] However, the loss of any lives is unacceptable; therefore, return of nuclear satellites must be performed in the safest manner possible. Therefore, it is imperative that all deorbited nuclear satellites be guided by active and reliable controls.

## 4.0 DEBRIS REMOVAL CONCEPT

Since current levels of debris are such that the density will increase even if no future space operations are conducted, it is necessary to implement a plan for the removal of a significant amount of debris. This need calls for a comprehensive removal plan which can reduce all types of debris. However, because of the variations in types and sizes of debris, a single removal system is not feasible. Therefore, two systems were considered as the solution to the problem of debris removal. These systems are an active system and a passive system, which remove debris greater than 20 centimeters in diameter and between 0.1 millimeter and 20 centimeters in diameter, respectively.

### 4.1 Active Removal Systems

According to Donald J. Kessler, the most effective method of debris management is the control of the large debris population. For this reason, various systems for larger debris management have been carefully examined. These systems target tracked debris and then actively dispose of it by various processes. The size of debris removed by an active system will be determined by the tracking capabilities and the expense of removal of each object. Current tracking abilities, shown in Figure 8, limit the range of debris which can be eliminated by active removal systems. Concurrently, the high cost of removal does not suggest attempting to actively retrieve debris smaller than can be tracked.

#### 4.1.1 Active Removal System Proposed

After extensive analysis, the proposed active removal system is the resuppliable roving system (RRS) consisting of multiple refueling modules and a configuration of ten roving vehicles (RVs), each equipped with eight deorbit devices. The primary purpose of each RV is to rendezvous with and capture debris, while the purpose of the deorbit device is to remove debris from orbit. Lastly, the refueling modules will be employed for refueling and resupplying the roving vehicles.

4.1.1.1 Roving Vehicle

The roving vehicle design, shown in Figure 11, is based on the Orbital Maneuvering Vehicle (OMV) designed by McDonnell Douglas [6:264] and shown in Figure 12. The inherent capabilities of the OMV include, rendezvous by teleoperated control, a replaceable propulsion module for easy refueling, a long-life power supply, and a high performance computer. Modifications were made to enable the RV to despin debris, to carry deorbit devices, and to attach these devices to debris. The robotic despin capabilities were adapted from the Proximity Operations Vehicle (POV) designed by Grumman Aerospace Corporation [6:421]. However, it is important to note that the RV does not have the capability to capture a tumbling satellite.

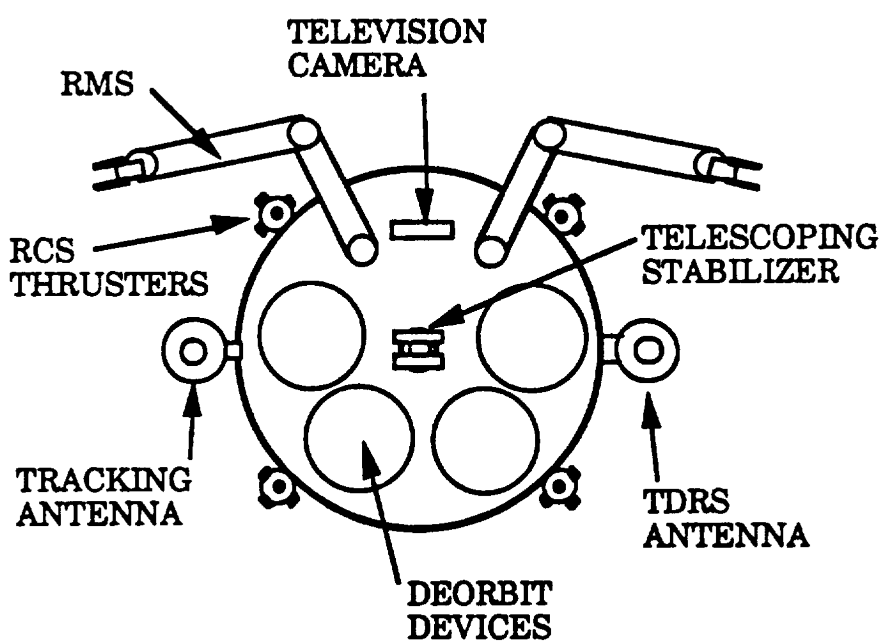


**Figure 11a. Roving Vehicle - Front View**

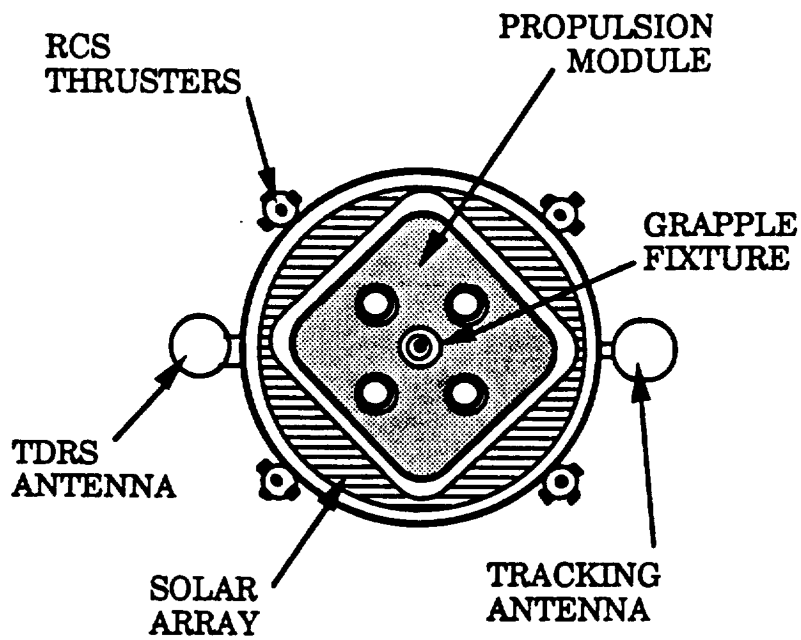


**Figure 11b. Roving Vehicle - Rear View**

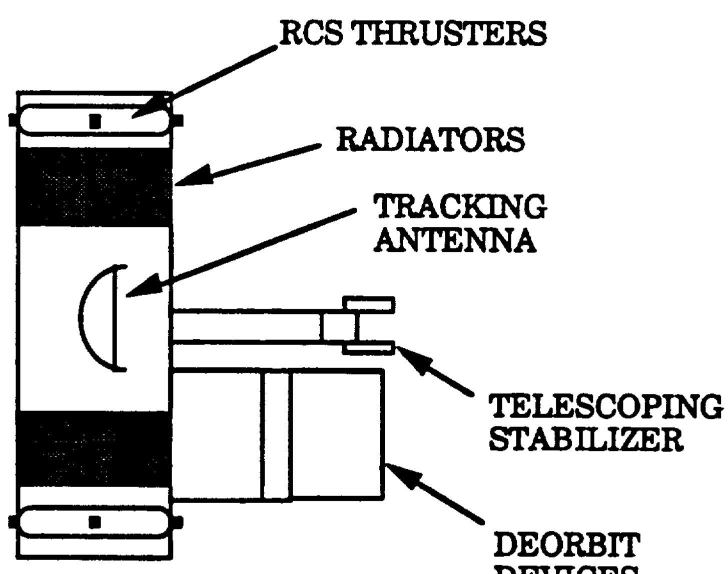


**Figure 11c. Roving Vehicle - Side View**

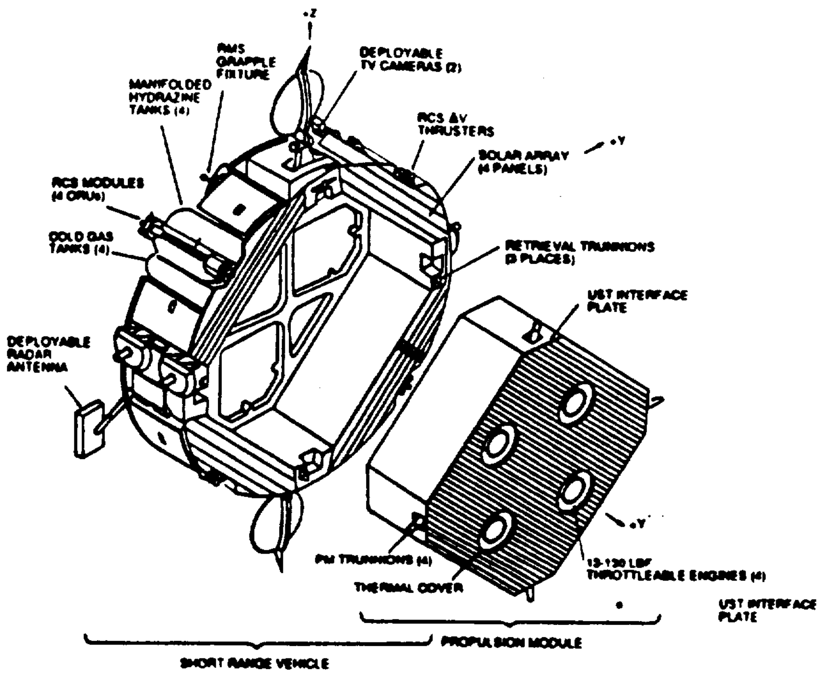


**Figure 12. Orbital Maneuvering Vehicle [16:26]**

The various RV subsystems were selected according to the processes described in Appendix A. The vehicle components and masses are listed in Table 2.

**Table 2. Roving Vehicle Subsystems and Masses**

| | |
|---|---|
| Propulsion Module | 5383 kg |
| 8 Deorbit Devices | 182 kg/each |
| Robotics from POV | 160 kg |
| OMV Subsystems & Base | 4085 kg |
| Total Mass | 11084 kg |

The RV's main subsystems are adapted from the OMV. These systems include: data processing system (DPS); guidance, navigation, and control (GNC); structures and mechanical; propulsion; and communication.

#### 4.1.1.1.1 Data Processing System

The DPS consists of the flight software and the general purpose computers. The software for the RV consists of standard "off-the-shelf" flight software that will require no adaptations for the purpose of debris removal. The primary purposes of the software will be to command flight control maneuvers, maintain redundancy management, control communication, execute the docking and rendezvous sequences, and to coordinate the effective use of all systems onboard the RV. The DPS will also employ a Freon loop system to reject the heat that will accumulate in the RV through the use of all the onboard systems. All elements of the DPS will be modular to allow for easy replacement in the event of a failure.

#### 4.1.1.1.2 Guidance, Navigation, and Control

The GNC system, controlled by the DPS system, consists of a rate determination system and a location determination system. The rate determination system includes two rate gyros to provide continuous attitude information, and an earth/sun sensor to provide updates or absolute attitude information. The location determination system consists of accelerometers, a global positioning satellite (GPS) receiver, and a tracking radar. The accelerometers provide provide information to the DPS for location determination while the GPS receiver will be used to update the state vector of the RV when necessary. The tracking radar is active for a range between 4.5 nautical miles to 35 feet from the target. After the target object is within 100 feet, the ground based controllers will begin remote control and rendezvous operations. All of the elements in the GNC system will be modular to allow for easy replacement in the case of a failure.

#### 4.1.1.1.3 Communication

The communication system is an integral part of both the DPS and the GNC system. For the RV, the communication system consists of S-band data links with the tracking, data, and relay satellite (TDRS) with the capability to perform data relay at variable rates. The RV will also be equipped with eight onboard cameras: two redundant docking cameras on the face of the RV, two deployable cameras, and two redundant RMS cameras per wrist for proximity operations. All of the major components will be modular and will allow for easy replacement if failures occur.

#### 4.1.1.1.4 Propulsion

The most significant subsystem on the RV is the propulsion system. The propulsion system consists of two major components, the propulsion module and the reaction control system (RCS). The RCS consists of 28 small hydrazine thrusters with a thrust of 15 pounds each in addition to 24 nitrogen gas thrusters with a thrust of 5 pounds each. The system is cooled through simple radiation to the environment and no active cooling for the propulsion system is required. The propulsion module consists of 4 variable thrust engines ranging in thrust from 13-130 pounds. The fuel is monomethyl hydrazine, and the oxidizer is nitrogen tetroxide. The total mass of the usable propellant is 9,000 pounds. The propulsion module and the RCS thrusters are replaced when the fuel supplies are exhausted, or the deorbit devices have all been distributed.

#### 4.1.1.1.5 Structures and Mechanics

The final major subsystem of the RV is the structural and mechanical system. The structural system consists of a bolted aluminum frame. Additional structural support, necessary to support the propulsion module, is provided by the trunnion and latch assemblies in the rear of the platform as shown in Figure 11b.

The mechanical subsystem involves the telescoping stabilizer in conjunction with the remote manipulator system (RMS) arms shown in Figure 11a. During proximity operations the stabilizer and the RMS will be ground controlled for approach and rendezvous. During refueling, the RV will use the RMS to remove supplies or replacement parts from the refueling module and attach the parts in the proper location on itself. All of the major mechanical components will be modular and easily replaced if necessary.

4.1.1.2 Deorbit Device

The deorbit device, shown in Figure 13, is the mechanism that will be attached to the orbital debris after rendezvous. The deorbit device will then force the space debris into an orbit that will enter the earth's atmosphere where the orbital debris and the deorbit device will burn-up during entry.

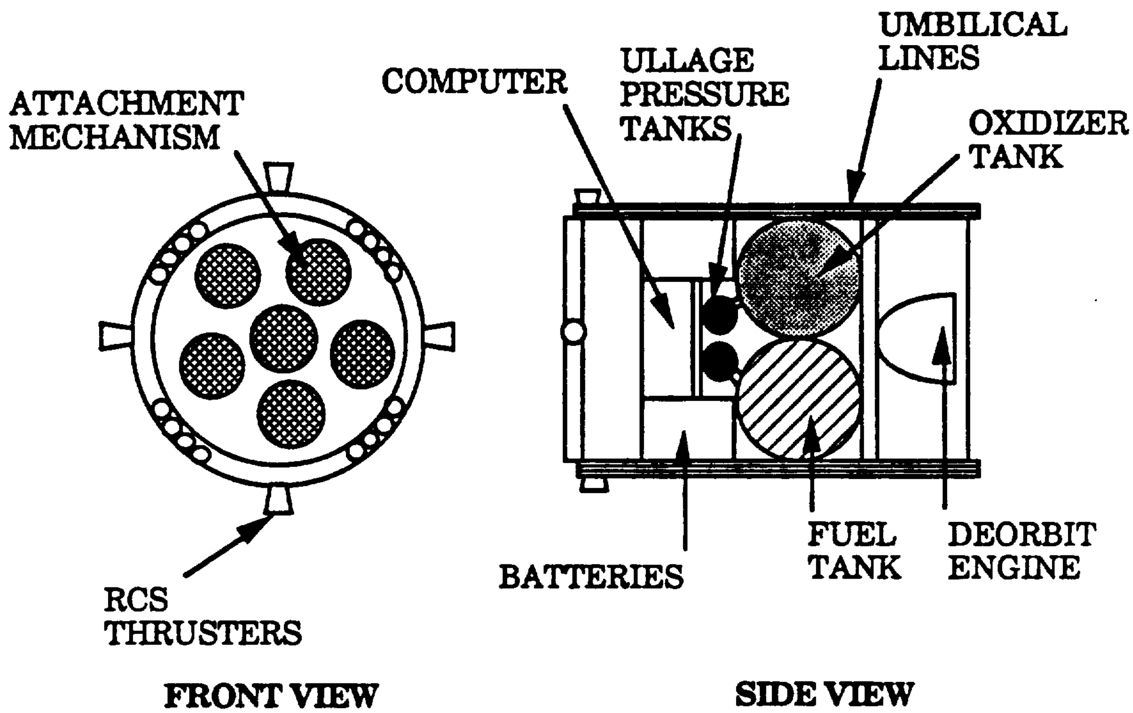


**Figure 13. Propulsive Deorbit Device**

The propulsive deorbit device subsystems required for control and propulsion were selected according to the procedures described in Appendix A and are listed in Table 3. Additionally, the propulsive abilities of the device were designed so that it can deorbit a satellite of at least 1000 kg, the average satellite mass in the operational range of the removal system.[12:70] Therefore, the deorbit device will generally be equipped with an engine similar to the one used in the Apollo lunar module ascent stage. When supplied with approximately 100 kg of propellant, this 3,500 pound force engine can deorbit a 1100 kg object from a 1000 km orbit to an elliptical orbit with a perigee of 100 km, as detailed in Appendix B. In addition, the time of this burn was calculated to be about 10 minutes, which is well within the operational limits of most rocket engines. If a piece of debris is significantly smaller or larger than can be removed by the designed deorbit device, the size of the tank, amount of propellant, and burn time may be adjusted as necessary.

**Table 3. Deorbit Device Subsystems and Masses**

| | |
|---|---|
| Inertial Measurement Units | 5 kg |
| Attitude Control | 31 kg |
| Reaction Control | 15 kg |
| Power (Pb-Acid Battery) | 1 kg |
| Computer | 4 kg |
| Structure | 6 kg |
| Propulsion (Lunar Ascent) | 20 kg |
| Fuel and Oxidizer Tanks | 100 kg |
| Total Mass | 182 kg |

As shown in Figure 13, the deorbit device is fueled by liquid monomethyl hydrazine and nitrogen tetroxide. The helium ullage pressure tanks are used to maintain pressure in the empty part of the fuel and oxidizer tanks as the engine is fired and the fuel and oxidizer are depleted.

The batteries are used to power the electric valves and the computer in the deorbit device. While the device is attached to the RV, it will be supplied with guidance information, electrical power, and thermal relief through the umbilicals shown in Figure 14. Deorbit device #2 is attached to the front of the roving vehicle while deorbit device #1 is attached to deorbit device #2. However, both devices receive supplies through the same umbilical cord. Once the device is attached to the targeted orbital debris, the umbilical will be released and the computer will begin independent operation of the guidance package necessary to complete the mission. The thermal conditioning will be performed through simple radiation to the environment; therefore, no dedicated, autonomous system will be required for heat rejection.

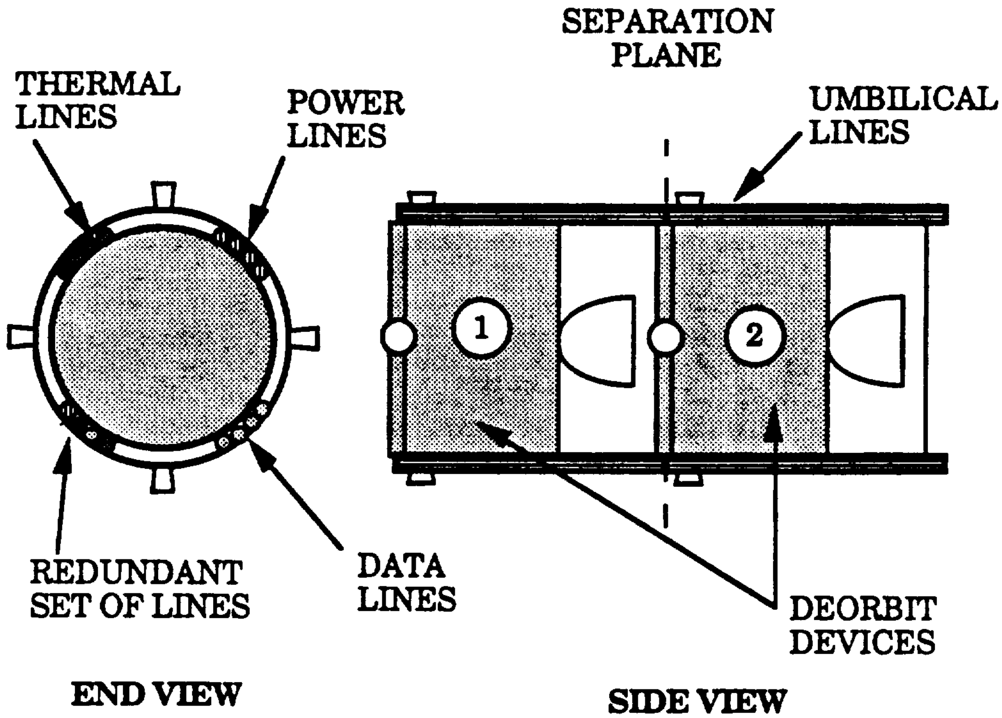


**Figure 14. Stacked Deorbit Devices**

4.1.1.3 Attachment Mechanism

As shown in Figure 13, the deorbit device will be attached to a piece of debris using the attachment mechanisms located at the end of the device. There are 6 attachment mechanisms per deorbit device, and contact with any one will be able to provide a sufficient bond to ensure the device will remain attached to the debris during the operation. The deorbit device will use the attachment mechanism to permanently adhere to the piece of orbital debris.

An enlarged view of the attachment mechanism is shown in Figure 15. This mechanism consists of four separate tanks and a mixing container. The first tank contains helium that will be used to pressurize the other remaining containers during the bonding process. The pressurization tank is connected to the other tanks through pipes controlled by electrically actuated valves. These valves will be controlled during the bonding process by the computer onboard the deorbit device. The second tank is the adhesive tank containing

FM-35, a commercially available adhesive produced by American Cyanimid [3:352]. The third tank contains the catalyst that will be used to cure the adhesive during contact. The recommended catalyst is an azidosilane because of the proven abilities of this substance as an adhesive promoter [8:9]. The final tank, the etching agent tank, contains the phosphoric acid that will be used to clean the surface of the space debris and prepare it for the adhesive process.

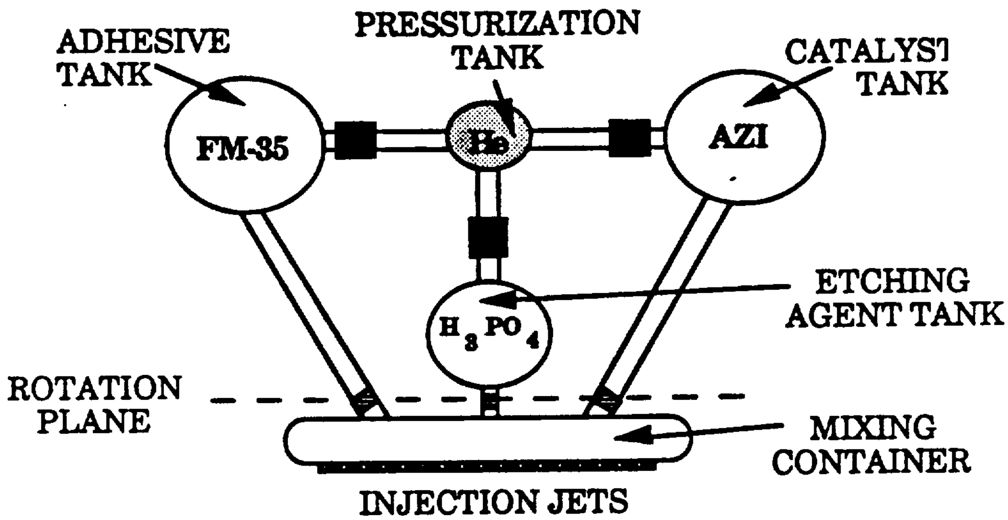


**Figure 15. Attachment Mechanism**

The attachment mechanism will be in a safe mode before contact between the deorbit device and the orbital debris to ensure that no accidental release of adhesive occurs. The adhesive process will begin once contact is made between the end of the deorbit device and the orbital debris. The injection jets and the mixing container will move about the rotation plane shown in Figure 15 to provide a flat surface for bonding. First, the etching agent will be dispensed to clean the surface of the orbital debris. Second, the adhesive and the catalyst will be injected into the mixing container and onto the prepared surface. The curing process should take about 10-15 minutes [9:118]. During this entire process, the pressurization tank maintains a constant pressure in the adhesive and catalyst tanks to ensure proper mixing. Once the debris has been permanently affixed to the deorbit device, the deorbit device and debris will be released from the roving vehicle.

#### 4.1.1.4 Refueling Module

The intent of refueling modules is to refuel and to resupply RVs. The refueling module, shown in Figure 16, will be launched directly from earth and will carry one propulsion module, eight deorbit devices, and any necessary replacement system modules. Because the roving vehicle will rendezvous with the refueling module, the refueling module will not need extensive robotics.

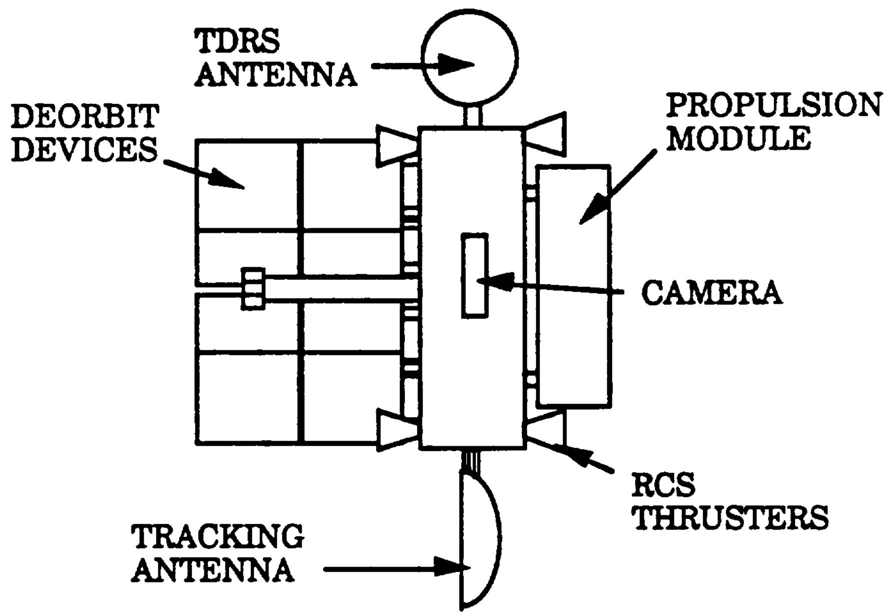


**Figure 16a. Refueling Module - Top View**

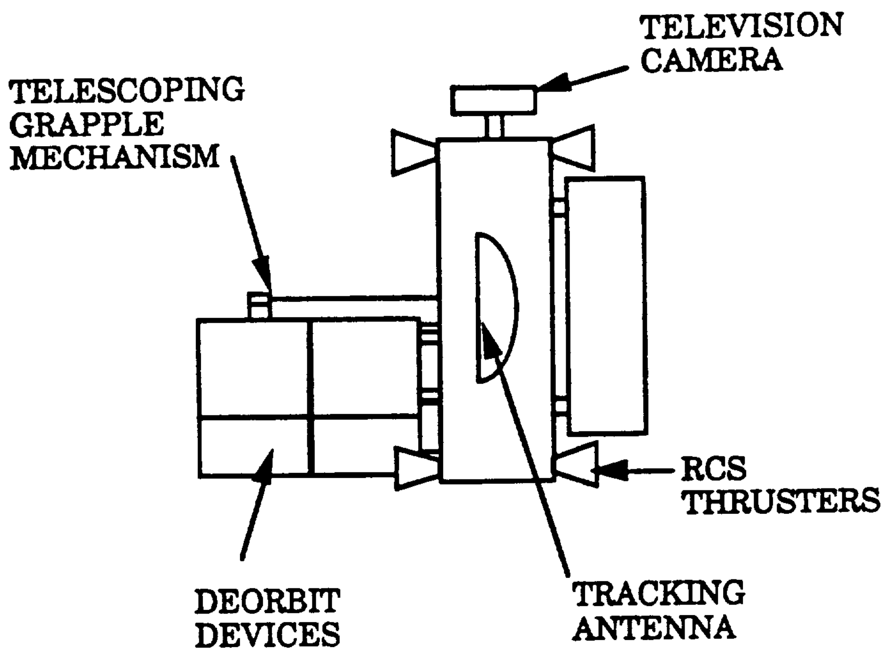


**Figure 16b. Refueling Module - Side View**

The subsystems required for the refueling module are listed in Table 4. These components were selected according to the guidelines explained in Appendix A.

**Table 4. Refueling Module Components and Masses**

| | |
|---|---|
| Inertial Measurement Units | 5 kg |
| Attitude Control | 31 kg |
| Reaction Control | 15 kg |
| Power (Pb-Acid Battery) | 10 kg |
| Computer | 4 kg |
| Structure | 20 kg |
| Communications | 10 kg |
| Propulsion (Lunar Ascent) | 20 kg |
| Fuel and Oxidizer Tanks | 150 kg |
| Propulsion Module | 5383 kg |
| 8 Deorbit Devices | 182 kg/each |
| Total Mass | 7104 kg |

The refueling module will be launched from the surface of the earth, and once in orbit, the roving vehicle will rendezvous with the refueling module. First, the roving vehicle will remove the deorbit devices from the refueling module and place them in the proper position on the front of itself. Second, the refueling module will remove the expended propulsion module from the back of the roving vehicle by grasping the grapple fixture with the grapple mechanism and activating the release switch. Finally, the refueling module will rotate and place the new propulsion module into the backside of the roving vehicle. Once these steps are complete, the refueling module will move away from the roving vehicle and will perform maneuvers necessary to place itself with the expended propulsion module into an entry trajectory for burn up in the earth's atmosphere.

#### 4.1.1.5 Vehicle Distribution

The apportionment of the roving vehicles was determined from the distribution of larger debris among the various inclinations and altitudes, as shown in Figure 5. Due to the significantly larger percents of debris at inclinations of 100°, 82°, 74°, and 66° two roving vehicles will be distributed in each of these inclinations, while only one roving vehicle will be placed in each of the two remaining inclinations, 91° and 32°. In order to service a wider range of debris, each vehicle is designed to service inclinations within ± 2° of its target inclination. In the orbits with two vehicles, the service inclinations will be separated by 4° in order to cover an even wider range of inclinations.

The concentration of operations will be between altitudes of 400 km and 1000 km because of the concentration of large debris between the altitudes of 175 and 1000 and the minimal affect of atmospheric drag above an altitude of 400 km. In addition, the roving system is capable of servicing higher altitudes if the need arises.

#### 4.1.1.6 Mission Scenario

Each of the roving vehicles' missions will vary for each set of targeted debris. However, each mission will involve five main steps: rendezvous, capture, deorbit, departure, and resupply, as shown in Figure 17.

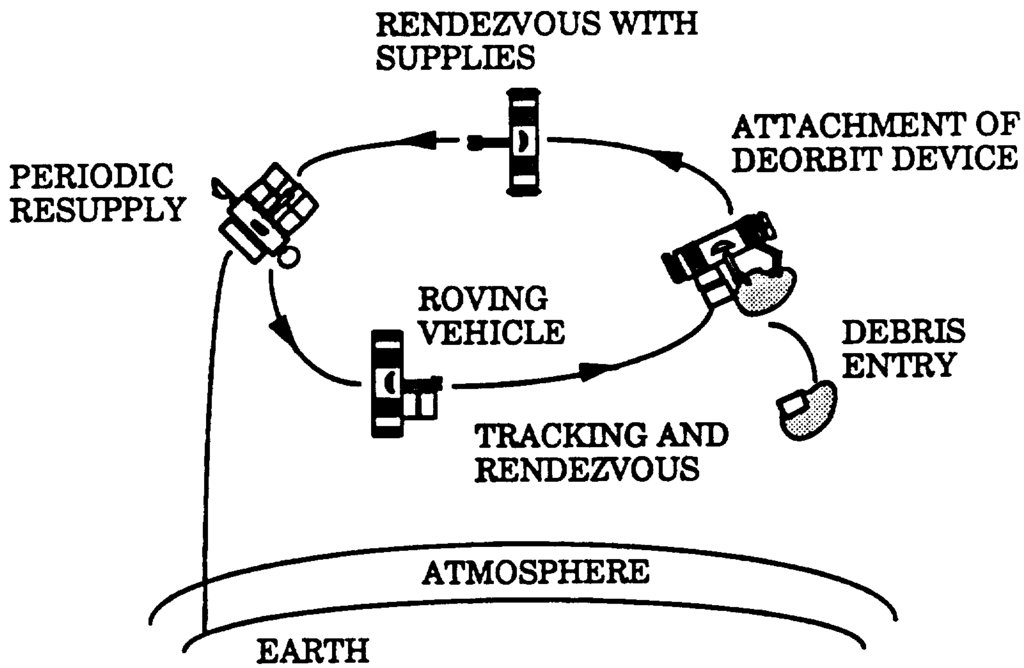


**Figure 17. Resuppliable Roving System Mission Scenario**

The first step of the mission is the rendezvous with the debris. The roving vehicle will depart its refueling orbit and follow a Hohmann transfer to the orbit of the first piece of debris to be captured. The next steps encompass the rendezvous with the debris, and its capture and despin. The RV, through teleoperated control, will rendezvous with the debris, attach to the debris with its grapple fixture, and, if necessary, the RV will then despin the object. The RV will then use its robotic arms to attach a deorbit device directly onto the debris. Once the deorbit device has been firmly affixed to the target debris, the deorbit device and debris will be released by the RV. Consequently, the deorbit device will begin the maneuvers necessary to place itself and the debris on a preplanned deorbit trajectory. Finally, the RV will proceed onto the next targeted object.

Once the RV has deposited all eight of its deorbit devices on various pieces of debris, it will place itself into a refueling orbit. Upon reaching the perigee of this orbit, the refueling module will rendezvous with the RV. The module will then begin procedures to furnish new deorbit devices to the RV and to replace the old propulsion module. Upon completion of these refueling and resupplying procedures, the refueling module will detach and return itself and the exhausted propulsion module to earth on a controlled trajectory. Finally, the RV will continue on to another set of debris and will repeat the process.

#### 4.1.1.7 Launch Vehicle

Before active orbital debris removal can begin, it is necessary to determine the proper launch vehicle for the system. Several candidate launch vehicles with their payload capacity to 500 km altitude are shown in Table 5. According to this information, the Delta 3920 is the best suited vehicle to launch the 7,104 kg refueling module. In addition, the Delta 6920 is best launch vehicle to carry the 11,084 kg roving vehicle.

**Table 5. Launch Vehicle Capabilities**

| Launch Vehicle | Payload Capability to Low Earth Orbit (kg) |
|---|---|
| Atlas Centaur | 5,000 |
| Delta 3920 | 9,100 |
| Delta 6920 | 11,300 |
| Titan 4 | 16,400 |
| Space Shuttle | 24,500 |

#### 4.1.1.8 Sample Mission

A sample mission was designed to provide actual numbers and to introduce several complicating factors to the mission design so as to validate the proposed system and its potential capabilities. This mission involves the removal of eight specified satellites that are currently orbiting the earth. Included in this mission are detailed $\Delta v$ and fuel mass analyses for each step of the 8 part mission.

The satellites, listed in Table 6, were selected from the 1982 TRW Space Log [14] according to several criteria. The most important requirement was that every satellite had to be in inclinations within the range of the system. Since plane changes are costly in fuel, the system is designed to service within 2° of its specific inclination. For this mission, a vehicle beginning from 97.5° inclination was selected; therefore, all debris was required to be within the range of 95.5° to 99.5° inclinations. The next major criteria in selecting a satellite was the eccentricity and altitudes of its orbit. In order to follow the assumption of circular orbits necessary for the proximity operations equations, Cauchy-Wilshire (C-W), only satellites in orbits with eccentricities less than 0.1 were selected. In addition, each of the satellites were evaluated according to altitude. The specified range of the RRS is 400 to 1000 km, but satellites also within 250 km of this range were chosen, as the RRS has the capability to remove them.

**Table 6. Satellites Selected for Removal in Sample Mission**

| Name | International Designation | Semi-major axis (km) | Inclination (deg) |
|---|---|---|---|
| - | 1965 21A | 862 | 99.0 |
| - | 1965 38A | 1120 | 98.2 |
| - | 1972 76A | 1130 | 98.7 |
| Tiros 10 | 1965 51A | 1250 | 98.4 |
| - | 1967 96A | 1190 | 99.2 |
| Meteor 1-28 | 1977 57A | 937 | 97.7 |
| SESP 74-2 | 1976 65C | 894 | 96.4 |
| - | 1980 10A | 635 | 96.9 |

Once a suitable group of satellites was chosen for the sample mission, the chronology of the mission was determined. Since even small plane changes require large amounts of fuel, the satellites were ordered first according to inclination with only a secondary emphasis on altitude. The schedule chosen for the mission is listed in Table 7. For this schedule, the total inclination change was 5.4° and the total change in altitudes was 1143 km. These variances are typical of the RRS missions. In addition, the total mission time was estimated to be 45 days by adding the synchronization times between each of the adjacent orbits, as shown in Table 7.

**Table 7. Sample Mission Chronology**

| Orbit | Int'l Desig. | Incl. (deg) | Semi-maj. axis (km) | Synodic Per. w/ adjacent orbit (days) |
|---|---|---|---|---|
| park | - | 97.5 | 500 | 7.83 |
| 1 | 1965 21A | 99.0 | 862 | 1.57 |
| 2 | 1967 96A | 99.2 | 1190 | 8.54 |
| 3 | 1972 76A | 98.7 | 1130 | 1.00 |
| 4 | 1965 38A | 98.2 | 1120 | 4.05 |
| 5 | 1965 51A | 98.4 | 1250 | 1.64 |
| 6 | 1977 57A | 97.7 | 937 | 12.81 |
| 7 | 1976 65C | 96.4 | 894 | 1.86 |
| 8 | 1980 10A | 96.9 | 635 | - |

The next step involved choosing transfer orbits and proximity operations. In order to minimize the fuel usage, Hohmann type transfers were assumed for the large scale transfers between orbits. Proximity operations were employed for rendezvous with the targeted satellite. To assist in the evaluation of these operations, Cauchy-Wilshire (C-W) equations had to be employed. By using these equations worst case fuel and time constraints for rendezvous were determined. The $\Delta v$ and fuel required for each of the transfers and rendezvous is shown in Table 8. The calculations performed for the sample mission are included in Appendix C.

**Table 8. Sample Mission Δv and Fuel Requirements**

| Transfer Orbit | Inclination change | Δ v (m/s) | Fuel Mass (kg) |
|---|---|---|---|
| 1 | 1.5 | 275.4 | 1018.1 |
| 2 | 0.2 | 164.6 | 568.7 |
| 3 | 0.5 | 69.7 | 255.5 |
| 4 | 0.5 | 63.6 | 224.7 |
| 5 | 0.2 | 67.3 | 221.8 |
| 6 | 0.7 | 177.1 | 487.6 |
| 7 | 1.3 | 169.1 | 429.2 |
| 8 | 0.5 | 150.3 | 358.2 |
| TOTAL | 5.4 | 1137.1 | 3563.8 |

The total fuel used in this mission was 3564 kg, which is only 87% of the fuel available; hence, the RRS is capable of performing as expected. Additionally, assuming successive missions of the same approximate duration as this mission allows each roving vehicle to complete up to seven missions annually, which corresponds to removing 56 pieces of debris per year. By retrieving such a large number of pieces of debris per year, the benefit would immediately be apparent, and each roving vehicle can remove a significant amount of debris can be removed within its estimated 10 year lifetime.

4.1.1.9 Cost

Several assumptions were made to accurately estimate the cost of the proposed active removal system. First, the cost of the OMV was modified from predictions originally made by Petro and Ashley. Their original predictions were modified to reflect the 10 RVs operating over 10 years for the proposed resuppliable roving system. Second, as described in the sample mission, it is expected that each RV would remove 56 pieces of debris per year and would be resupplied by 7 refueling modules during the year. Therefore, 5600 pieces of orbital debris would be used during the 10 project life. Third, the cost of the refueling module was estimated to be 1/10 that of the RV. Finally, the cost estimates include research and development, and launch vehicles, but do not include facilities for support operations or the replacement of faulty parts.

**Table 9. Cost of Proposed Active Removal System**

| Mission Element | Total Cost | Cost Per Piece of Debris |
|---|---|---|
| Roving Vehicle | | |
| Refueling Modules | | |
| Deorbit Devices | | |
| Overall Cost | | |

### 4.1.2 Other Active Removal Systems Studied

In order to better understand the selection process used in choosing the resuppliable roving system, it is important to examine the alternative active removal systems considered. These systems included a resuppliable orbital base system, a direct removal system, and a laser beam unit system.

#### 4.1.2.1 Resuppliable Orbital Base System

One active system considered, a resuppliable orbital base system (ROBS), is a modification of the resuppliable roving system previously described. ROBS requires six orbiting refueling bases in addition to the ten RVs with eight deorbit devices each. The main purpose of each refueling base is to store propulsion modules and deorbit devices for the RVs.

##### 4.1.2.1.1 Orbiting Refueling Base

The orbiting refueling base, shown in Figure 18, is designed to store excess deorbit devices and propulsion modules; therefore, it mainly consists of structure. However, the base does have the ability to perform all procedures necessary for the refueling and resupplying of the roving modules, which includes replacing propulsion modules and attaching deorbit devices to the RVs. This capability requires that the base have extensive robotics and telecommunications abilities. Specifically, the base is designed to hold 4 propulsion modules and 36 deorbit devices.

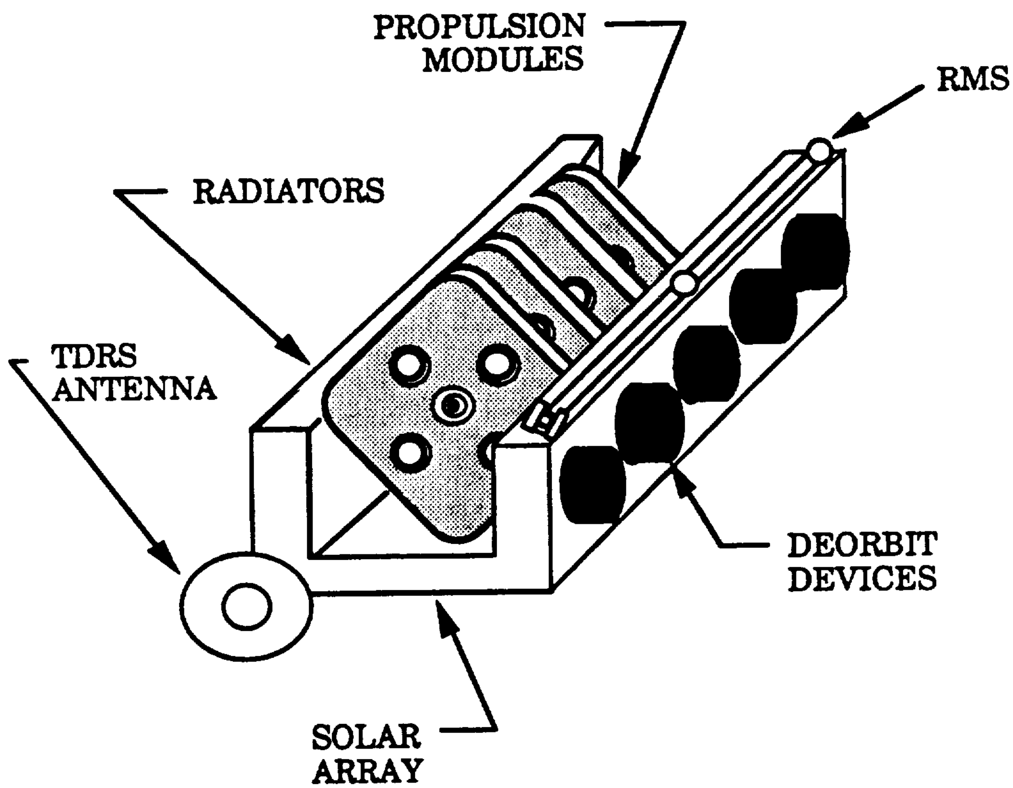


**Figure 18. Orbiting Refueling Base**

4.1.2.1.2 Vehicle Distribution

The distribution of the roving vehicles will be the same as for the roving system. Additionally, a refueling base will be placed at each of the six inclinations, where RVs are placed, to eliminate the need for substantial plane changes. Since the RVs will concentrate their operations between 400 km and 1000 km, the base will be placed in an elliptical orbit in this altitude range.

4.1.2.1.3 Mission Scenario

The mission scenario for ROBS, shown in Figure 19, is basically the same as that described for the roving system. Unlike the roving system where each RV is refueled and resupplied by an earth launched refueling module, the RV in the orbiting base system is refueled and resupplied by the base in its inclination. Rather than launching to each RV, a package of two propulsion modules and eighteen deorbit devices would be launched to the base, where they would be stored. Upon distribution of all eight deorbit devices, the RV will rendezvous with the base where its propulsion module is replaced and eight additional deorbit devices are attached. After the RV departs for another mission, the base would attach a deorbit device to the exhausted propulsion module, and it would be deorbited on a controlled trajectory.

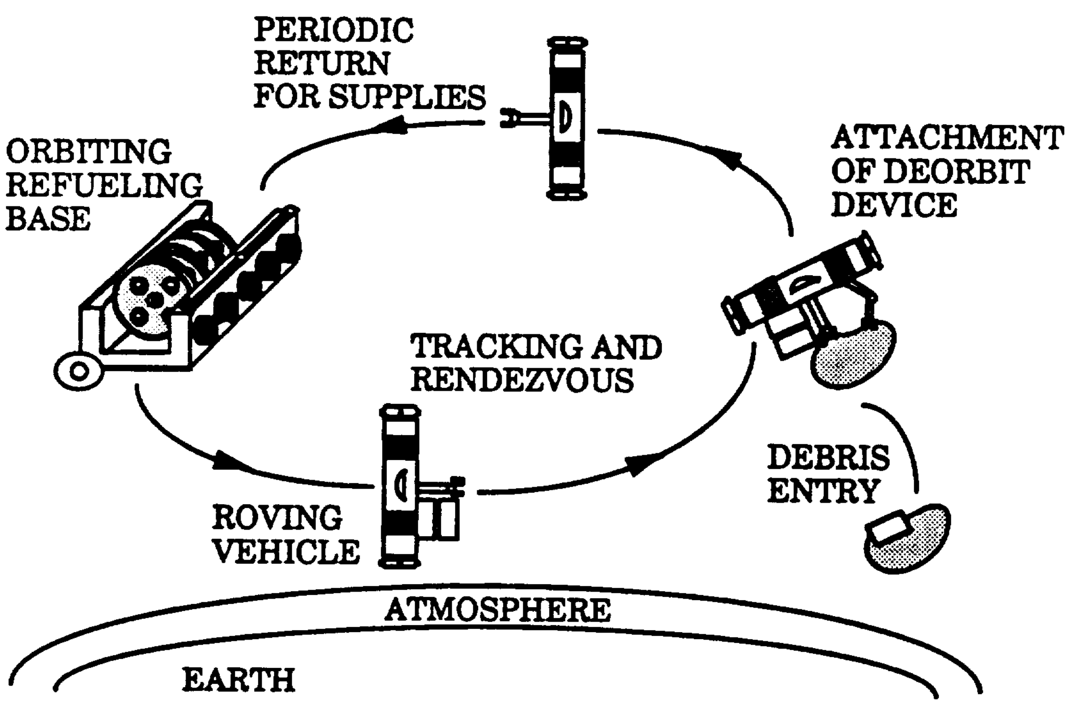


**Figure 19. Resuppliable Orbiting Base System Mission Scenario**

#### 4.1.2.1.4 Advantages

Since the resuppliable orbiting base system is simply an alteration of the roving system, the ROBS has the same basic advantages. The only difference is the further reduction of launches required. By using a base for refueling and excess supply storage, the launches have been cut in half, and therefore the cost per mission is reduced.

#### 4.1.2.1.5 Disadvantages

As in the system advantages, the ROBS disadvantages are basically the same as those for the RRS. There are, however, additional disadvantages due to the addition of the base. These include a much higher initial investment due to the cost of the six bases. Another disadvantage is the difficulty in upgrading the refueling systems when onorbit refueling becomes available. In addition, the replacement of damaged systems modules on the roving vehicle is more difficult than for the RRS since the bases will be supplied less often. Therefore additional modules would need to either be stored in orbit, or the RV would be forced to wait until another fuel supply was launched to the base with the replacement equipment.

Because of the enumerated advantages of the RRS over the ROBS, this system was not selected.

### 4.1.2.2 Direct Removal System

Another active removal system studied, the direct removal system involves rendezvous, capture and deorbit of a single piece of debris. The process of removal would be done by a earth launched single rendezvous and return vehicle.

#### 4.1.2.2.1 Single Rendezvous and Return Vehicle

The single rendezvous and return vehicle (SRRV), shown in Figure 20, is based on the POV proposed by Grumman Aerospace Corporation. The POV is designed with the robotic and telecommunications capabilities needed to rendezvous capture and despin space objects. However, due to POVs limited fuel supplies, additional fuel has been incorporated in order to deorbit the complete system. The design weight for the POV is 500 kg, and the necessary extra fuel is estimated to be 80 kg.

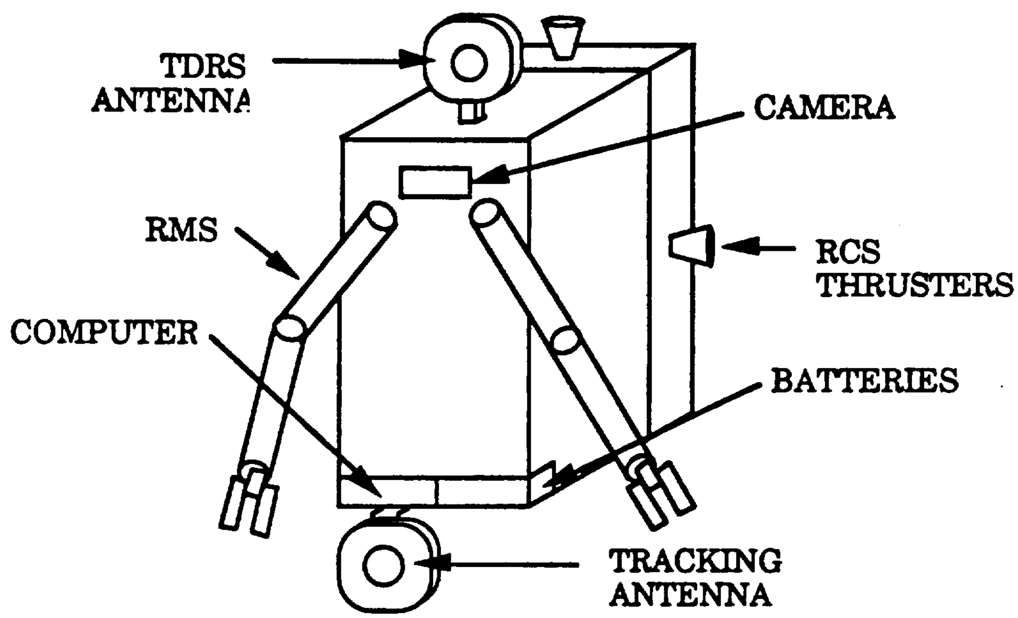


**Figure 20. Single Rendezvous and Return Vehicle**

4.1.2.2.2 Vehicle Distribution

The direct removal system has no actual distribution of vehicles. Since each SRRV retrieves only one piece of debris, it will be directly launched to an orbit convenient for rendezvous with the targeted debris.

4.1.2.2.3 Mission Scenario

The mission scenario, shown in Figure 21, for the SRRV will vary for each vehicle and each piece of debris. However, each mission will involve four main steps: launch, rendezvous, capture, and deorbit. For each mission, the SRRV will be directly launched from earth to an orbit necessary for rendezvous with the targeted debris. Next, the SRRV will rendezvous with the debris, capture it, and, if necessary, despin it. Finally, the SRRV will return to earth with the debris on a controlled, preplanned trajectory. The amount of time and fuel for the mission depends completely on the exact orbit of the debris and the launch site.

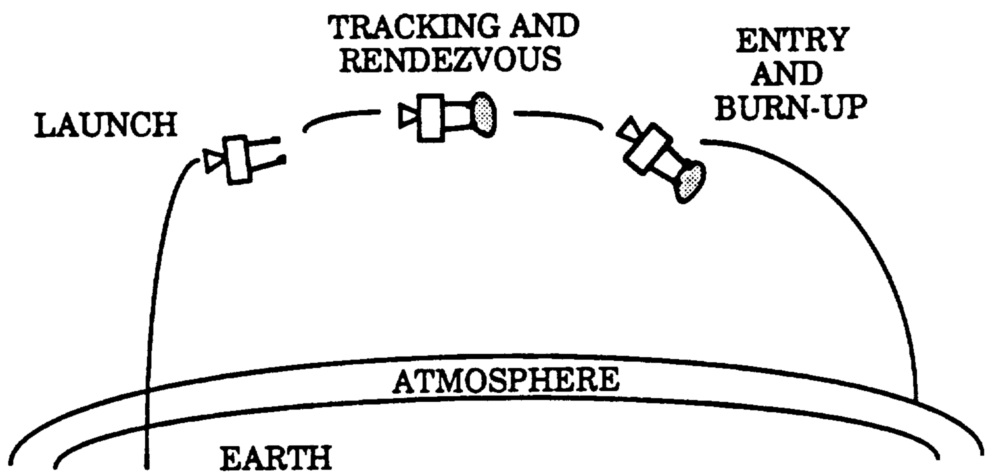


**Figure 21. Single Rendezvous and Return Vehicle Mission Scenario**

4.1.2.2.4 Advantages

The DRS has a few distinct advantages over the proposed system. These advantages include a lower initial cost because of the high cost of launching and maintaining permanent systems. In addition, the removal vehicle is significantly lighter than the resuppliable roving systems. Finally, the weight of each SRRV is low enough that the Minuteman II Missile assisted by six Castor motors can successfully launch a single SRRV to lower altitude orbits.

4.1.2.2.5 Disadvantages

The main disadvantage of the DRS is its inefficient use of resources. By completely destroying a removal vehicle with the removal of a single piece of debris, the total amount of equipment required would be extreme. Additionally by requiring a separate launch for each piece of debris removed, the cost per piece of debris is significantly increased.

Due to the expected high costs of removal per piece of debris as well as the inefficient use of materials, launch vehicles, and funds, the direct removal system is not a reasonable solution.

4.1.2.3 Laser Beam System

The final active system investigated is the laser beam system. This system consists of ten laser beam units which target and track debris and then remove it by imparting a change in velocity using a low power laser beam.

4.1.2.3.1 Laser Beam Unit

The design of the laser beam unit (LBU), shown in Figure 22, is based on the principles of using a laser beam to impart a change of velocity to an object using photon pressure. The system is also designed to track targeted debris once it is within 10 km, and then verify that the object is the target by its orbit, size, or other distinguishable characteristics. The system is also equipped with the propulsion systems necessary for minor orbit changes in order to come within firing range of the targeted debris.

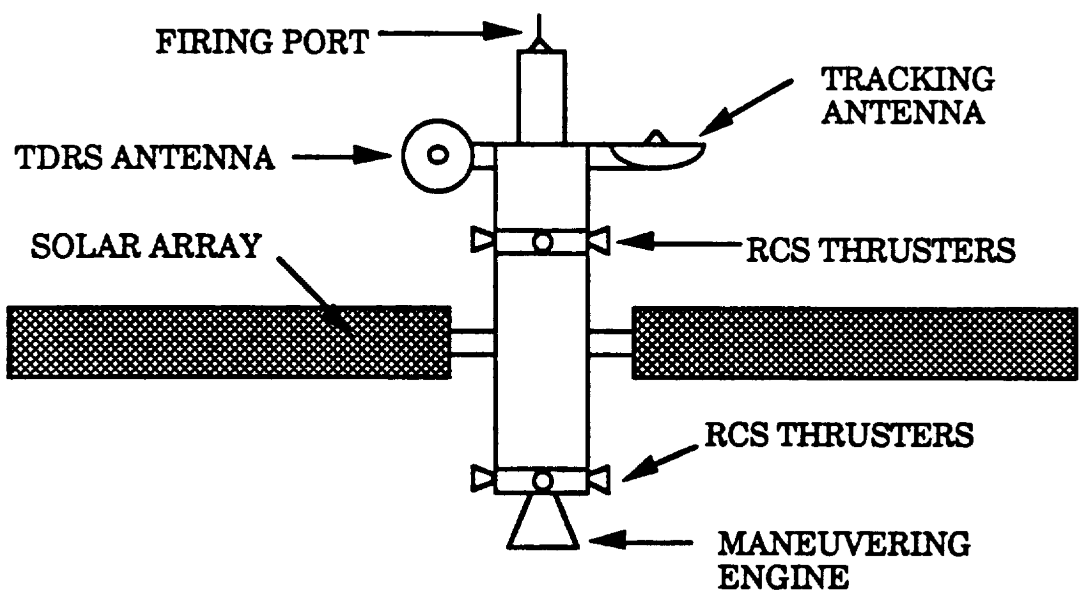


**Figure 22. Laser Beam Unit**

A Ruby laser was chosen because of its relatively high efficiency, long life, and moderate power outputs.[5:162] The power output is important when studying the effect of the laser on the debris. All lasers will both impart momentum and vaporize some of the material. However, the design requirement to not create additional debris makes it essential that the vaporization of material be minimized. By reducing the output power of the laser, the damage to the debris, and thereby the gaseous debris from vaporization are reduced.

#### 4.1.2.3.2 Unit Distribution

The division of the laser beam units will again be based on the distribution of the larger debris. However, since the LBUs are effective over a long range, and, unlike the RVs and SRRVs, they need not rendezvous with debris, fewer units are necessary in each inclination. Therefore only 6 units are necessary, where one unit operates in each of the six inclinations of 100°, 91°, 82°, 74°, 66°, and 32°.

#### 4.1.2.3.3 Mission Scenario

Specific mission scenarios, similar to the one shown in Figure 23, for the LBUs will vary according to the specific placement of the targeted debris. Moreover, the missions would be similar to those for the RVs with the exception that an estimated of 20 to 30 pieces of debris could be removed without refueling. More debris can be removed since the LBU does not have to achieve the same speed as the targeted debris in order to impart a $\Delta v$ to it. The only requirement is that the relative velocity of the debris be low enough that the debris stays within range long enough for the LBU to impart the necessary $\Delta v$ to it.

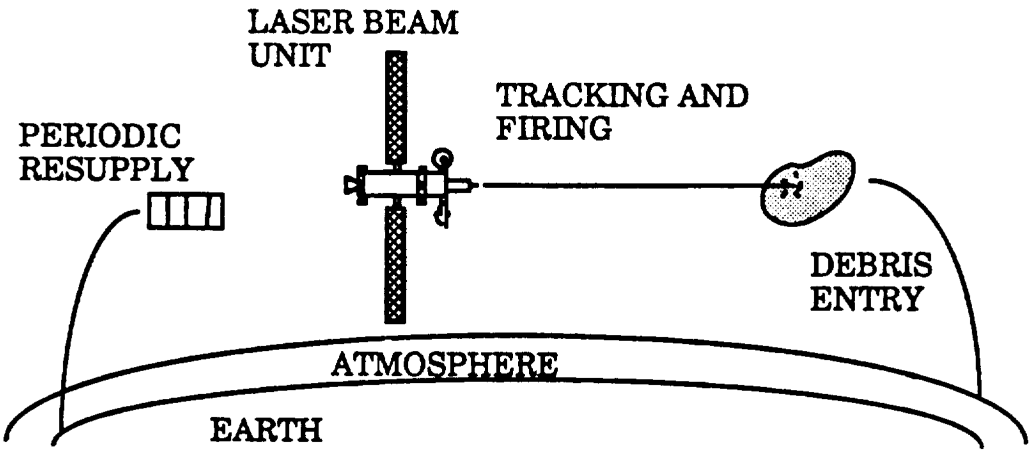


**Figure 23. Laser Beam System Mission Scenario**

4.1.2.3.4 Advantages

Several of the advantages of the laser beam system are the disadvantages of the other active removal systems. These include the ability to dispose of tumbling satellites and to eliminate the need for rendezvous with debris.

4.1.2.3.5 Disadvantages

Although advantages of the laser system are numerous, the disadvantages are cause for great concern. The main disadvantage is the time required for an average size of debris. The change in velocity by applying photon pressure using a 1 KJ laser is related to the mass of the debris by

$$\Delta v = 3.335 \times 10^{-6} / m \text{ [17:F95]}$$

where

$\Delta v$ = change in velocity per pulse and
$m$ = mass of debris (kg).

Therefore, if the laser is operated at a rate of one pulse per second, the total time required to impart a $\Delta v$ of 30 m/s to a 1000 kg satellite is approximately 285 years. The other option of using a higher energy beam to impart a $\Delta v$ by vaporization is not a viable solution because it results in significant amounts of microscopic debris which does not meet the requirements for the removal system.

Because of the time required for deorbit of a single piece of debris, this system is unrealistic for a viable removal system.

## 4.2 Passive Removal Systems Studied

Disposal of the smaller debris will be performed by passive systems. These systems were designed to sweep that part of the debris population that cannot be managed with the active system. A passive system is intended to eliminate debris ranging in size from 0.1 mm to about 10 cm in diameter. Because this debris is undetectable by ground based radar, passive systems must remove small debris through random occurrences. Therefore, each system must have at least a 10 km range of effect to have an appreciative effect on the debris environment within 10 years.[15:5.14]

After applying the selection criteria to the candidate passive systems, it was determined that no effective and safe passive system was currently available. However, further technological developments may make one of the studied passive removal systems viable or may present new methods of passive removal. The decision not to propose a passive removal system can be justified by the relative unimportance of small debris with respect to the hazards posed by large debris. Specifically, although the smaller debris constitutes the largest percentage of the debris population, it is easier to protect against and hence less hazardous. In addition, according to Kessler's debris growth model previously described, removal of small debris is less effective in controlling the debris environment than removal of large debris.

### 4.2.1 Umbrella Satellite System

The first passive system of debris removal which has been studied is a umbrella satellite system. This system would consist of ten, 1 km diameter, deployable umbrella satellites, that would remove small debris via random collisions.

4.2.1.1 Deployable Umbrella Satellite

The deployable umbrella satellite (DUS), shown in Figure 24, must be constructed of a shielding material which will not leave secondary debris when impacted by small debris.

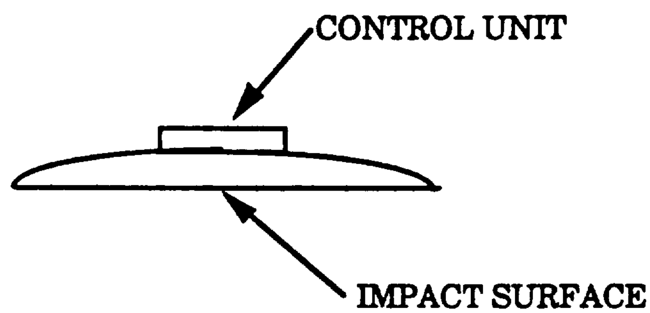


**Figure 24. Deployable Umbrella Satellite**

These satellites were designed to be equipped only with the minimum hardware necessary for station keeping and minimal communications. However, at this time, no shielding techniques are available which leave no secondary debris. Therefore, the specific mass of a DUS has not been determined.

4.2.1.2 Satellite Distribution

The placement of the satellites is determined by the distribution of small debris among the various altitudes and inclinations. The inclination distribution is the same as for the large debris, but the small debris is more evenly spread among the altitudes between 400 and 1000 km. Two umbrella satellites would be put in slightly elliptical orbits in each of the most cluttered inclinations of 66°, 74°, 82°, 100°, and one unit would be placed in each of the two remaining inclinations, 32°, and 91°. The elliptical orbits would be designed so that they overlap so that all altitudes are covered. The orbits must also be designed such that the satellites will not collide with any active operations.

#### 4.2.1.3 Mission Scenario

All debris removal by the umbrella satellites is completely random. As shown in Figure 25, debris which impacts a umbrella satellite will imbed in the satellite. Once it has completed its removal of small debris, the satellite and all of the imbeded debris will return to earth on a controlled trajectory.

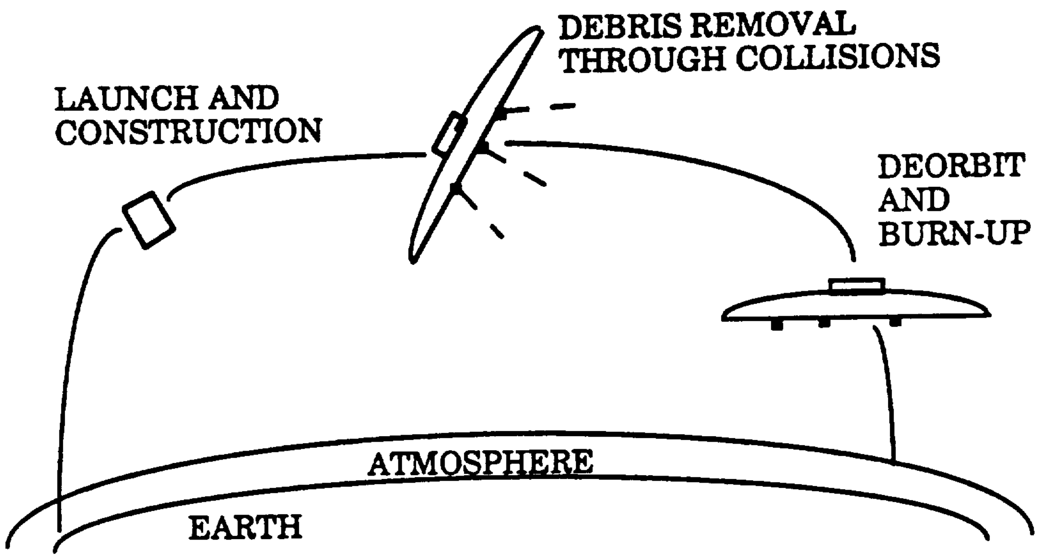


**Figure 25. Umbrella Satellite System Mission Scenario**

#### 4.2.1.4 Advantages

The only advantage of the DUS is that it requires a minimal amount of subsystems. Since most of its life will be spent in a single orbit, the only subsystems needed would be station keeping and attitude adjustment equipment and a small communication system.

#### 4.2.1.5 Disadvantages

Unfortunately, the DUS has several major shortcomings. Since a large part of the system will consist of an outer shell, a shielding material must be used that can withstand hypervelocity impacts without producing any secondary debris, and currently this type of a shield does not exist. Another major disadvantage of the umbrella satellite is its deployed size. A structure with a 1 km diameter, would be very heavy, expensive to launch, and difficult to control.

### 4.2.2 Foam Ball System

The second passive system investigated is a constellation of foam balls, which also capture debris through random collisions. In order to have the necessary effective range of ten kilometers, this system consists of ten 1 km diameter foam balls at various altitudes and inclinations.

#### 4.2.2.1 Foam Ball Satellite

The foam ball satellite, shown in Figure 26, is designed to be constructed of a light weight durable material. This material must be able to withstand impact from debris without leaving secondary debris. These satellites are not designed for any station keeping or communications abilities; therefore, the weight and size of the vehicle is completely determined by the material chosen. However, there is no material which currently meets the requirements, of no secondary debris and lightweight.

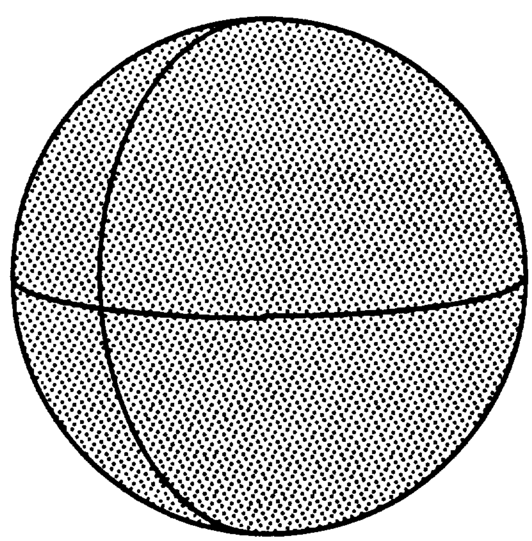

**Figure 26. Foam Ball Satellite**

4.2.2.2 Satellite Distribution

Like the umbrella satellite system, the placement of the foam ball system is determined by the distribution of small debris among the various altitudes and inclinations. Additionally the satellites must be positioned so that they will not collide with active operations. Since the removal method of the foam ball system is the same as that of the umbrella satellite system, the distribution of satellites would be identical.

4.2.2.3 Mission Scenario

Similar to the umbrella satellite, the foam ball satellite will capture debris by random collision. Upon completion of its mission, or once the impact of debris has repositioned the satellite in an orbit considered hazardous to other active operations, the foam ball satellite would be retrieved by an active removal system, as shown in Figure 27.

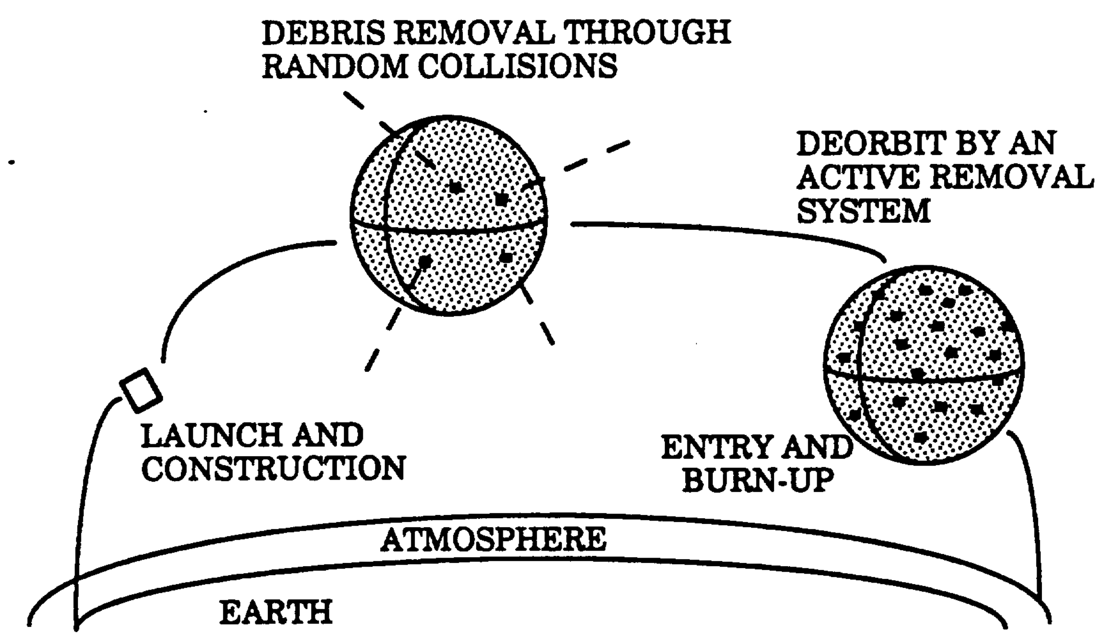


**Figure 27. Foam Ball System Mission Scenario**

#### 4.2.2.4 Advantages

The advantage of the foam ball is its complete lack of subsystems. This system is the simplest of the passive retrieval systems because it has no mechanical systems. This lack of subsystems significantly reduces the initial costs as well as maintenance costs.

#### 4.2.2.5 Disadvantages

The foam ball system may be a very cost effective solution, but currently no material has been shown to withstand high velocity without creating secondary debris. Additionally, the lack of control on the satellite allows it to stray from its initial orbit, and possibly place active operations in danger of collision. The final disadvantage is the size of each satellite. Since a passive system must have an effective range of 10 km per altitude, it is necessary to make the foam balls very large, which would increase the likelihood of collision with active payloads.

### 4.2.3 Smart Laser Beam System

The final passive system under consideration, the smart laser beam system, is a modification of the active laser beam system previously described. Since the laser beam units each are effective for a range of approximately 2 km, this system requires only five units to meet the range criteria. Each unit removes debris by imparting $\Delta v$ to identified debris encroaching upon its 2 km sphere of influence.

#### 4.2.3.1 Smart Laser Beam Unit

The smart laser beam unit is simply a modified version of the active removal laser unit, shown in Figure 22. The main differences are the sensor required for tracking debris and the computer capabilities necessary to identify debris that is not trackable from the earth. The smart LBU is designed with a LADAR sensor with a range of 10 km and a Ruby laser with an approximate range of 2 km.

Since the occurrence of debris within the range of the LBU is random, the system does not actively pursue debris. However, the directed firing toward detected debris is a very systematic and structured action. Therefore, the system does not remove debris completely by chance, as do the other systems.

The basis of the laser system is to use directed energy to impart a $\Delta v$ on a particle. The magnitude of the velocity change varies with the size of the object, the intensity of the laser, and the firing time. The application of photon pressure with only minimal the vaporization of material is very slow to impart a $\Delta v$ as shown by equation 1; however, the use of higher power lasers to vaporize the material does not meet the design system requirement of no additional debris.

The sensor chosen is a LADAR (laser detection and ranging) which uses the photons of light collected by its aperture for detection of objects. This sensor has a range of 10 km for detecting objects as small as one micron in diameter [11:216]. In addition to its detection abilities, the LADAR is capable of imaging an object within range.

The identification of detected objects as debris is an important requirement of the smart laser beam system. This necessity requires sophisticated data processing capabilities, such as an artificial intelligence system for debris recognition.

#### 4.2.3.2 Unit Distribution

Like the other passive systems, the smart LBUs are positioned according to debris distribution. However, fewer units are required than for the foam ball system and umbrella satellite system because the range of operation is much farther. Therefore, one smart laser beam unit will be placed in each of the most cluttered inclinations of 100°, 82°, 74°, 66°, and 32°.

#### 4.2.3.3 Mission Scenario

The disposal of debris by the smart LBU is not completely random like the other passive systems. As shown in Figure 23, the smart laser will fire upon detected and identified debris. The $\Delta v$ imparted on the object will eventually cause the debris to enter the atmosphere.

#### 4.2.3.4 Advantages

One primary advantage of the smart laser beam unit is its relatively small size. Specifically, this low mass makes the use of Minuteman II Missiles possible, which may significantly reduces the expense of the launch vehicle. In addition, due to the smart LBU's sensing and identification capabilities, the smart LBUs may be placed in orbits where active operations pass within the range of the LBU, provided they are not in danger of collision. Another advantage is the unit's larger range of operation than the foam ball and the umbrella satellite. Because of this range, fewer satellites are needed to have an equivalent effect on the orbital environment.

#### 4.2.3.5 Disadvantages

The main disadvantage of the smart laser beam system is the firing time required to deorbit debris. A 1 cm sphere of aluminum at 0.027 grams requires over 4 min for a 30 m/s $\Delta v$. Additionally, even the low energy photon pressure will vaporize some of the material and hence will contribute to the microparticulate debris population. This vaporization basically renders the system ineffective by changing each piece of debris into a more destructive cloud of debris.

### 4.2.4 Charge Repulsion System

The charge repulsion system is based on the observations that objects in space develop a negative electrical charge due to solar activity and emissions. Space-borne radiations, such as photons, electrons, protons, ions, cosmic rays, and x-rays,[28:2] result in charges up to 10-20 keV in the dark and up to hundreds of volts in the sun to develop on space objects.[13;4] The charge repulsor system consists of a constellation of ten repulsor satellites each of which creates a 1 km electric field resulting in the required total effective range of ten kilometers.

4.2.4.1 Repulsor Satellites

The repulsor satellite (RS), shown in Figure 28, is designed to create a positive electric field approximately one kilometer in diameter, where the field repels negatively charged debris. The change in momentum imparted on a piece of debris is dependent on the mass of the debris, on the angle of incidence at encounter, and on the magnitude of the charge of the debris and of the field. The generated electric field is normal to the surface of the generator; therefore, a change in velocity will be in the relative direction of approach. Since this direction is random, the exact momentum exchange is difficult to estimate.

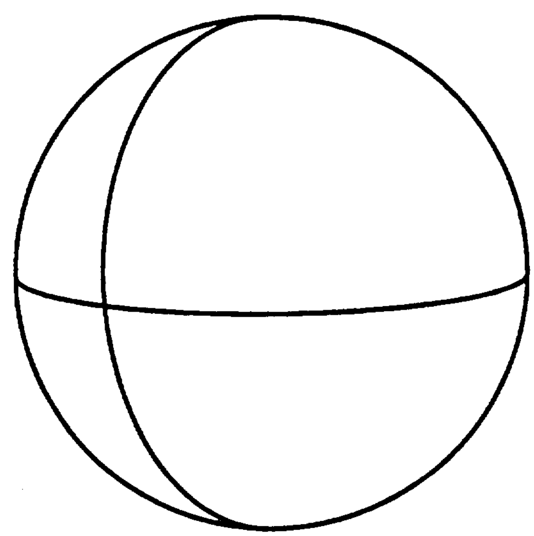

**Figure 28. Repulsor Satellite**

The main components of the repulsor satellite are the field generator and the power source. Because of the uniformity of the field created and the low power requirements, a Van de Graaff generator was chosen as the field generator.

4.2.4.2 Satellite Distribution

Like the first two passive systems described, the placement of the charge repulsor system is determined by the distribution of small debris among the various altitudes and inclinations. Each satellite is designed to have an effective range of 1 km. Therefore, only ten satellites are required. Two repulsor satellites will be put in each of the most cluttered inclinations of 66°, 74°, 82°, and 100°, and one unit will be placed in each of the two remaining inclinations, 32° and 91°.

4.2.4.3 Mission Scenario

The removal of debris by the repulsor satellite is through random repulsion of debris. As shown in Figure 29, this force will slow the debris and hence cause its eventual entry into the atmosphere.

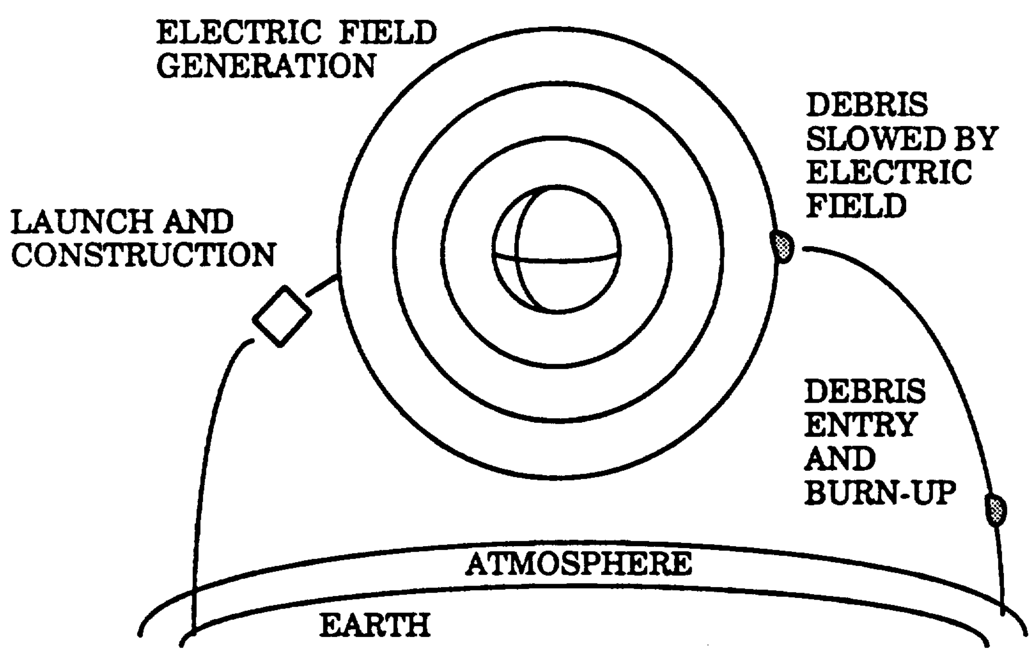


**Figure 29. Charge Repulsion System Mission Scenario**

#### 4.2.4.4 Advantages

The charge repulsor system has several advantages over the other studied passive systems of removal. One main advantage is the availability of the technology to create such a field. A uniform and reliable electric field of moderate strength can easily be produced without significant power requirements. Additionally, the repulsor would never be in direct contact with debris, and hence would not to produce fragment debris like the umbrella and foam ball satellites or vaporization debris like the laser system.

#### 4.2.4.5 Disadvantages

The restricting disadvantage of the charge repulsor system is its limited effectiveness. Although a negative charge will develop on space objects, only dielectrics will retain the charge once the imposing force is reduced or removed.[2:4] All conducting materials, such as aluminum, will dissipate any obtained charge very rapidly.[2:4] Therefore, the repulsor system would only be effective on the dielectric materials, and hence would not affect a significant portion of the orbital debris population. Furthermore, because of the adverse effects on computer systems, spacecraft are being designed with fewer dielectrics in order to reduce charges. Therefore, the percent population of debris affected by a charge repulsor system will continue to diminish.

## 5.0 DEBRIS PREVENTION CONCEPTS

Although the proposed removal program will have a profound effect on the reduction of present levels of orbital debris, more work needs to be done to ensure that future debris levels can be controlled. A coordinated effort on an international level will be necessary to encourage the use of uniform design standards that will curtail the growth of additional debris. Despite the increased mission costs associated with these changes, modification of mission hardware and space practices to prevent orbital debris is far more economical than the addition of an entire mission to recover debris from a previous mission. Even employing methods to reduce production of small debris is far less expensive than shielding against such debris or dealing with any damage resulting from such debris.

### 5.1 Prevention Techniques

If the growth of orbital debris is to be restricted, future hardware design for launching and space operations must implement one or a combination of prevention techniques. Some of these techniques are improved shielding, the addition of deorbit devices, and modifications of existing systems.

#### 5.1.1 Improved Shielding

Although meteorite shielding is already incorporated into current payload design, more stringent requirements are needed for improved shielding against debris impacts. Moreover, the shield must eliminate creation of secondary debris caused by such impacts. Aluminum louvers returned from the Solar Maximum Satellite provide evidence of the need for a reliable impact bumper system. Figure 30 depicts the penetration of a 0.14 mm thick outer louver by a piece of orbital debris. Hypervelocity tests indicate that the 0.52 mm-diameter hole was most likely the result of a collision with a 0.114 mm-diameter particle traveling at 10 km/s.[7:41] In Figure 31, the spray pattern on a second Solar Max louver consists of multiple craters due to secondary debris. This particular louver was located 3 mm behind the plate in Figure 30 and was in direct contact with the material that passed through. A multi-wall structure, such as the dual louver bumper, can be very effective in absorbing debris impacts without endangering spacecraft operations and further degrading the space environment with the generation of secondary debris.

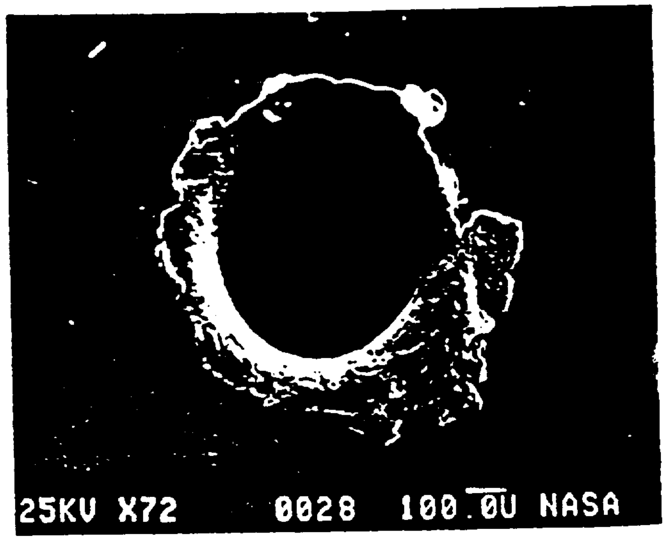


**Figure 30. Damage to a Front Louver of Solar MAX [10]**

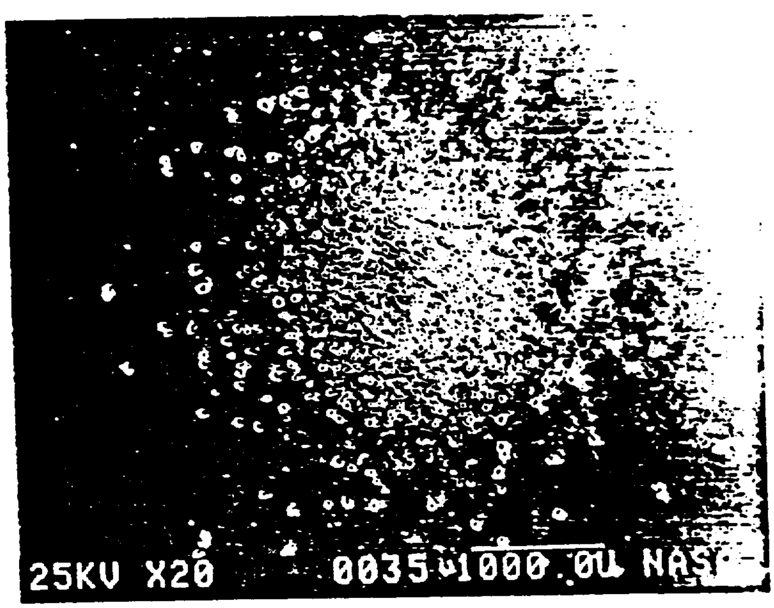


**Figure 31. Spray Pattern on a Back Louver of Solar MAX [5:89]**

Recent breakthroughs in shielding concepts have resulted in multi-layer bumper systems that can effectively withstand impacts from larger objects than previous shields and significantly reduce secondary ejecta. Burton G. Cour-Palais has conducted extensive hypervelocity research and has designed a multi-layer bumper shield, shown in Figure 32, that can withstand large debris impacts and considerably minimize secondary debris.[3] Other research includes the corrugated single or multi-layer bumper shield depicted in Figure 33. This shield was proposed and designed by Dr. Bill Schonberg from the University of Alabama-Huntsville specifically for the minimization of secondary debris.[9]

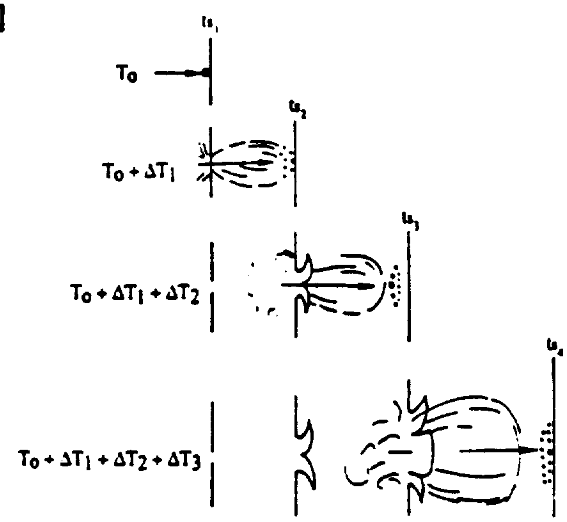


**Figure 32. Mulit-layer Bumper Shield [3:10]**

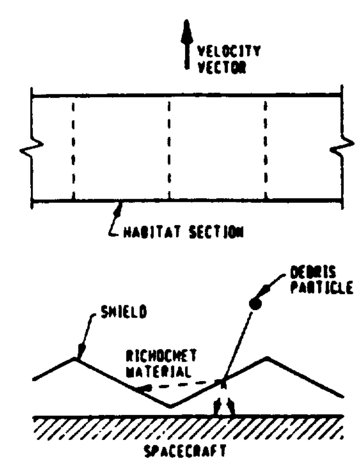


**Figure 33. Corrugated Bumper Shield [10:12.26]**

Even though these concepts produce some secondary material, the resulting amount of debris after collision is greatly minimized. Therefore, incorporation of these bumper systems into future spacecraft designs would significantly reduce the creation of smaller debris particles.

### 5.1.2 Onboard Deorbit Devices

The most important means of preventing further contamination of the space environment is by deorbiting a payload, its launch vehicle, and other mission related hardware upon completion of their useful life. Deorbit procedures can be achieved in a variety of ways that include the use of propulsive devices and/or the reliance on natural phenomena.

#### 5.1.2.1 Drag Deorbit Balloon

The foremost natural phenomenon that affects orbital decay of payloads is atmospheric drag. Below altitudes of 500 km, the density of the earth's atmosphere provides a significant retarding force to earth satellites. Even though the effects of atmospheric drag are enhanced or diminished depending upon mass and cross-sectional area, virtually all objects below 500 km enter the earth's atmosphere within a few years. The Long Duration Exposure Facility (LDEF), for example, decayed from an initial 400 km altitude to 300 km in 6 years and was in danger of entering the earth's atmosphere when it was retrieved by Space Shuttle Atlantis. Once above 500 km, the effects of atmospheric drag are considerably reduced. For example, a satellite in a circular orbit of 1,000 km is expected to remain in orbit for 1,000 years or more.[7:176] However, if the area of a spent satellite or rocket body could be increased, it would deorbit at an accelerated rate. This increase in area is the principle behind the drag balloon. This proposed deorbit device would be included as part of a mission payload and would be inflated once the working payload reached the end of its useful life, as shown in Figure 34. The balloon can also be used on rocket casings and other mission equipment that have shorter operational lifes.

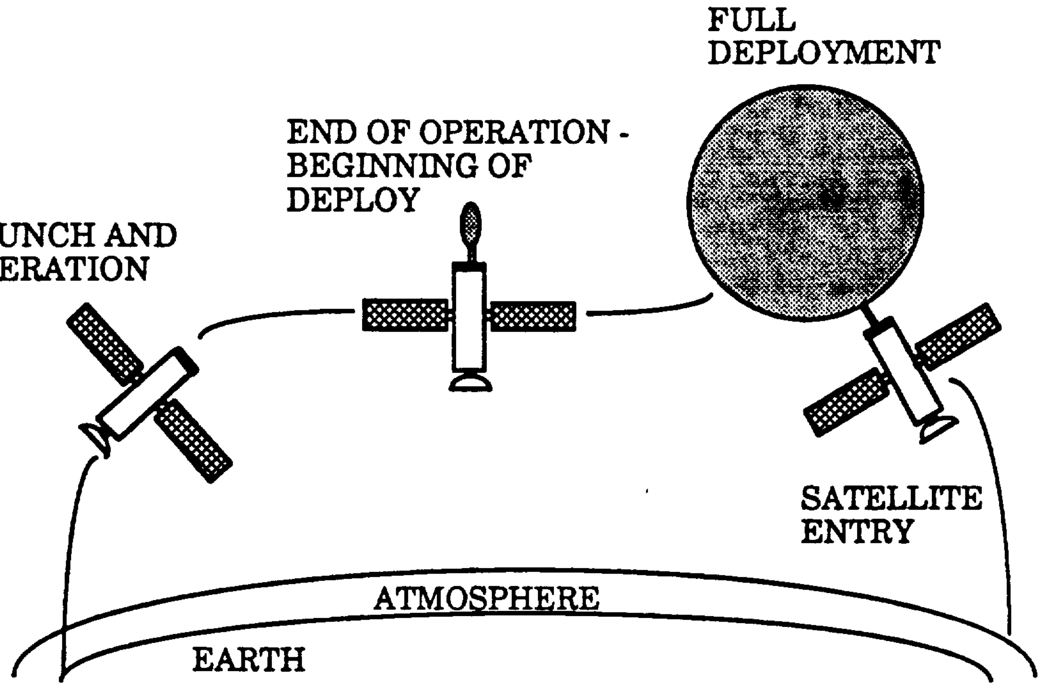


**Figure 34. Space Debris Prevention Using Drag Deorbit Balloon**

Because this concept relies on the earth's atmosphere, it has several disadvantages. Unlike an active deorbit, the orbital decay caused by the drag balloon is fairly slow as illustrated in Figure 35. Perhaps the biggest disadvantage of the balloon, is its ineffectiveness at high altitudes. As altitude increases past 500 km, atmospheric drag effects decrease dramatically. As a result, the balloon's surface area must increase in order to have the same effectiveness at the higher altitude. Accordingly, the balloon is only practical for altitudes up to 500 km, and is ineffective above altitudes of 750 km.[7:180] The variation of balloon size and mass as a function of altitude appear in Figures 36 and 37, respectively.

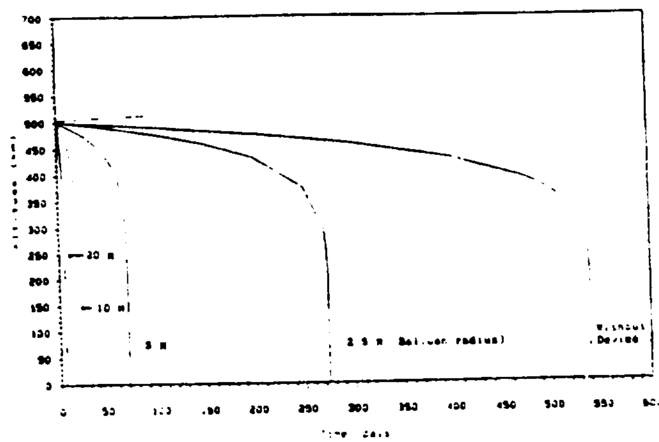

**Figure 35. Decay Time with and without Balloon Drag [7:177]**

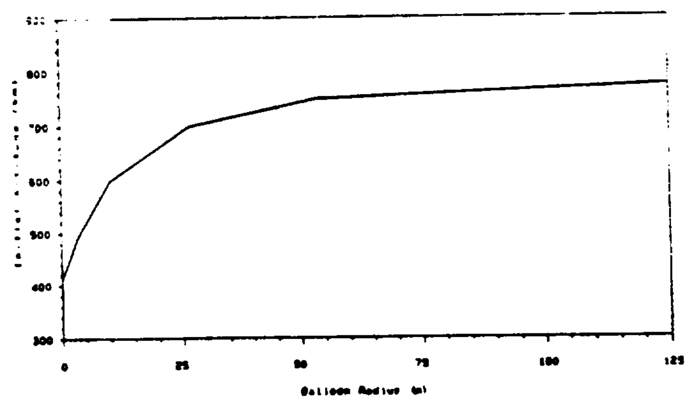

**Figure 36. Balloon Radius Radius Required for 90-Day Decay [7:178]**

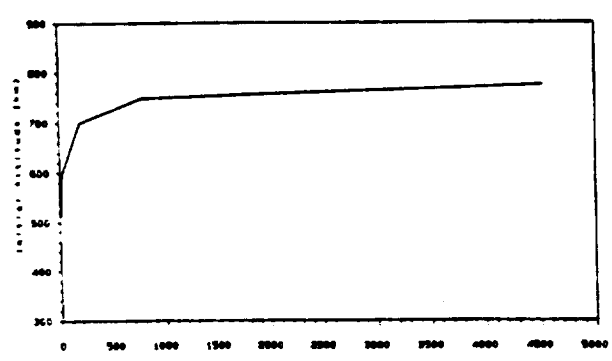

**Figure 37. Balloon Mass Required for 90-Day Decay [7:178]**

5.1.2.2 Deorbit Engine

For payloads that are deployed above 500 km, small deorbit engines can be used to deorbit the payload upon completion of its useful life. For proper operation, the device would remain safely inert for the entire operational life of the spacecraft and then be used for deorbit upon conclusion of active operations, as shown in Figure 38. Such a system, depicted in Figure 13, would naturally increase the payload weight, but like the drag balloon, it is still much less expensive than active retrieval.

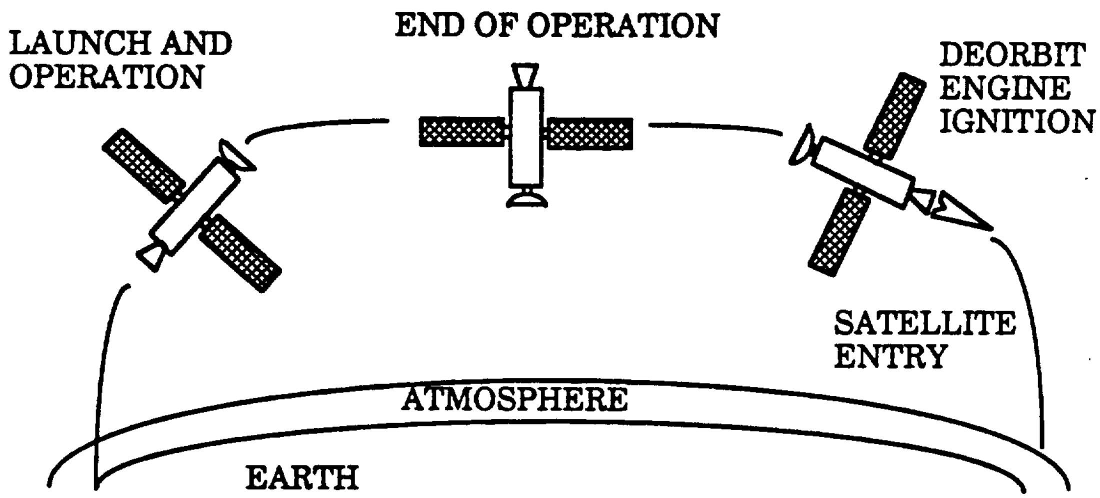


**Figure 38. Space Debris Prevention Using Deorbit Engine**

Although the deorbit engine poses a weight penalty for an individual mission, it is the most versatile of the deorbit devices. It not only allows for quick deorbit maneuvers, but it is also effective at any altitude. Unlike the drag balloon, its mass is relatively constant despite increasing altitude as shown in Figure 39.

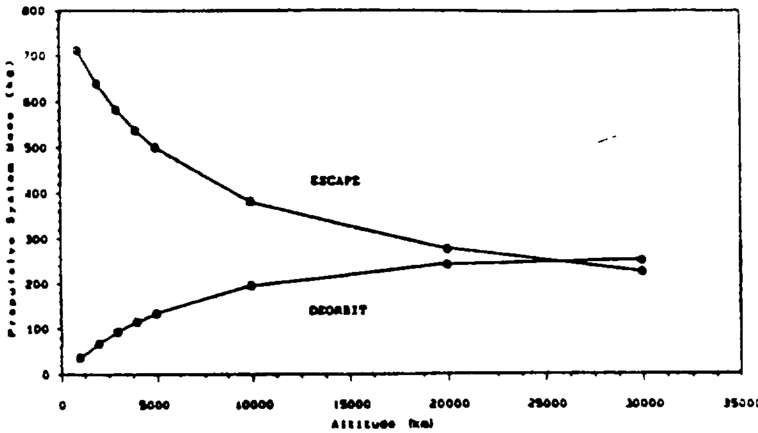


**Figure 39. Mass Required for Propulsive Deorbit Device [7:179]**

5.1.2.3 Additional Station Keeping Fuel

Perhaps one of the more practical methods of deorbiting useless payloads is to add a small percentage of fuel to the station keeping motors. Normally, station keeping motors are used periodically to reboost satellites to higher altitudes or to make minor orbital adjustments. However, adding more fuel would enable the station keeping motors to act as deorbit engines once the useful life of the satellite has ended. The mass penalty caused by additional fuel is shown in Figure 40. Other than this weight penalty, the addition of fuel requires no additional engines or other deorbit devices and is therefore, relatively cost efficient.

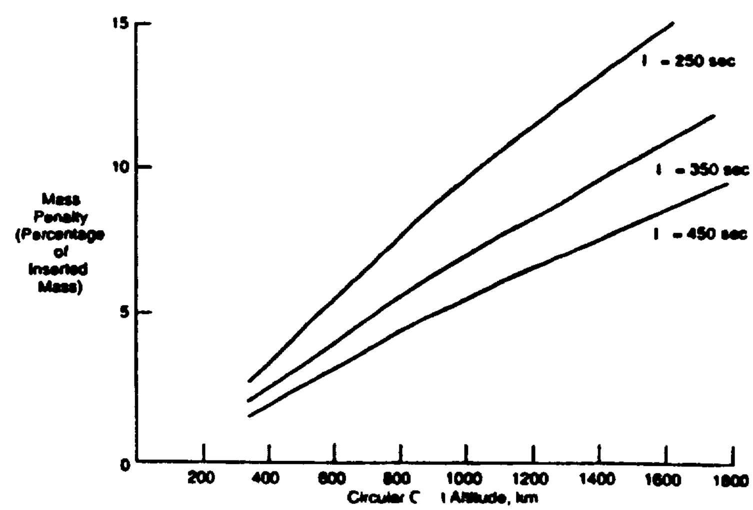


**Figure 40. Mass Penalty for Additional Fuel [10:5.9]**

### 5.1.3 Design Alterations

Much of the man-made debris currently in orbit is due to satellite breakups, upper stage explosions, and debonding of synthetic materials. Because the use of an active or passive method of debris disposal is extremely costly, it will be much more practical and cost effective if steps are taken to prevent the deterioration of payloads. Specifically, these preventive steps would consist of redesigning or modifying current mission hardware.

#### 5.1.3.1 Rocket Redesign

As the communications industry continues to thrive, more and more high altitude satellites are being launched every year. Not only do the spent rocket casings clutter the space environment, but they are subject to a high probability of failure. One such rocket, an Ariane third stage, exploded in 1986 and injected over 200 trackable objects and countless undetectable items into orbits ranging from 450 to 1450 km. Redesign of problematic rocket stages could prevent catastrophic failures, as well as, minimize the severity of breakup and the number of explosion fragments. The main design change necessary is to arrange for the depletion of all pressurized propellants.

#### 5.1.3.2 Separation Mechanism Redesign

Perhaps one of the most difficult design problems of the multistage rocket is to produce a clean stage separation. Most of the current launch vehicles are referred to as 'dirty' rockets because they make use of explosive stage connecting bolts to separate rocket stages. This procedure results in a debris cloud that, if at orbital speeds and altitudes, adds to the orbital debris environment. Therefore, payload separation mechanisms need to be redesigned so that shrapnel is either contained or prevented altogether.

#### 5.1.3.3 ASAT Altitude Limitations

Another major contributor to the orbital debris population has been the series of ASAT tests that have been conducted since the late 1960s. The Soviets alone have produced over 550 trackable items since they exploded their first satellite in 1968.[7:122] The best method of stopping ASAT tests would be to put an international ban on them; however, this is not likely. Instead, intentional explosions should be required to take place at low altitudes to allow fragments to enter in a short period of time.

#### 5.1.3.4 Development of Durable Bonding Agents

Another pending problem concerning satellite deterioration is the degradation of bonding agents. Bonding agents are commonly used in paints and protective coatings on most payloads. When exposed to atomic oxygen and the harsh thermal effects in space, these bonding agents breakdown, causing paint and coatings to fleck. These small bits of debris become microparticulate projectiles that are capable of causing extensive damage to spacecraft. Therefore, to prevent an increase in these hypervelocity objects, alternative bonding agents need to be developed so that the hazards of the space environment can be safely endured.

#### 5.1.3.5 Battery Redesign

A much less publicized cause of debris is explosions of batteries due to dead shorts. These explosions can be prevented by simply adding minimal electrical protection circuits to all batteries.

## 5.2 Prevention Treaties

Currently, States are not liable for damage caused by orbital debris; therefore, there exists no incentive to avoid generating it. In order to ensure that the orbital environment is protected, two issues, liability for damage by and required removal of debris, need to be addressed. Since currently no international agreements or laws call for the control, reduction, or elimination of orbital debris, treaties must be created or altered in order to encourage the cooperation of all countries in the protection of the orbital environment.

### 5.2.1 Liability Treaty

Compensation for damage caused by debris in outer space will never be an adequate substitute for preventing the generation of orbital debris. However, some legal mechanism is necessary so that States and private owners can recover losses that occur as a direct result of orbital debris. In space law, this function falls under the jurisdiction of the Convention on International Liability for Damage Caused by Space Objects (Liability Convention).[1] The Liability Convention is a United Nations (UN) sponsored treaty-resolution that was entered into force on October 9, 1973. Under the Liability Convention, States are absolutely liable for damage caused by their space objects to the surface of the earth or to aircraft in flight.

Two significant facts concerning the Liability Convention should be noted. First, negotiations for the Liability Convention were triggered by concerns over the possible harm to persons and property on earth from atmospheric entry of space objects. From the United States perspective, for example, the fundamental purpose of the negotiations was to provide compensation for damage resulting from these hazards. Second, in order to ensure the drafting of a treaty that was satisfactory to all parties, negotiators in the United Nations Committee on the Peaceful Uses of Outer Space (UNCOPUOS) specifically did not address several questions thought to be "relatively exotic" at the time.[8:129] One such question was the risks posed by orbital debris. As a result, the Liability Convention does not adequately address the issue of damage to persons or property in outer space.

The limitations of the Liability Convention lie in its ambiguous definition of "damage" and "space object". Article I of the Convention states :

> The term "damage" means loss of life, personal injury or other impairment of health; or loss of or damage to property of States or of persons, natural or juridical or property of intergovernmental organizations.

While damage to persons or property is included in this provision, damage to the outer space environment is not. Since no compensation is available for environmental damage, launching states cannot be held liable for the mere presence of debris in outer space. Therefore, launching states have no legal incentive to avoid the generation of orbital debris.

Article I also states, "The term 'space object' includes component parts of a space object as well as its launch vehicle and parts thereof."[1] This description, however, is not specific enough to include orbital debris. Although it does include operational debris, this definition excludes inactive satellites, fragmentation debris, microparticulate matter, and litter.

Since the Liability Convention does not adequately include damage caused by orbital debris or compensation recourse for damage caused in space, an amendment to the Liability Convention is proposed to include damage to space operations from any identifiable orbital debris, where orbital debris shall be defined as: "Any object in outer space deemed to be valueless, as evidenced by an absence of operational control, and includes inactive payloads, mission-related equipment, payload remnants, and microparticulate matter." In addition, identifications shall be determined by catalogued tracking or by distinguishing characteristics such as mission names, country insignias, or any other unique features.

### 5.2.2 Removal Treaty

Existing international agreements on space exploration do not address the issue of orbital debris removal. It is proposed that negotiations begin on a United Nations sponsored agreement requiring the timely removal of all payloads and related mission hardware at the completion of their useful lives. According to the Registration Convention of 1976, all spacecraft must be registered with the UN previous to launch.[2] In order to determine the maximum time before required removal, a projected useful life for the spacecraft and any related equipment would be required at the time of registration, and the launching State would be required to remove each separate object within two years of its estimated life.

If the launching State fails to remove their equipment from orbit after the two year time period, it is proposed that a policing agency be established within the UNCOPUOS to consider disciplinary and enforcement actions. The launching country would be given an opportunity to appeal the deorbit of a payload. If the country can show the value and use of allowing the payload to remain in orbit, the agency would be empowered to extend the payload lifetime for a specified length of time. In the event of no appeal or insufficient evidence of usefulness or activity, the agency could contract an independent source for the disposal of the payload and related hardware in question. Resulting costs and expenses would then be billed to the launching State.

In the event that the launching State refuses to pay for removal costs, the policing agency could consider other disciplinary options. Specifically, it could direct UNCOPUOS to refuse allocation of GEO slots to that State, refuse registration of future launches, or request the restriction of technology transfers to that State's space agency, thereby isolating it from the international space community.

## 6.0 MANAGEMENT REPORT

The management structure of STRES, Inc. is summarized in Figure 41. Tasks are assigned as follows. The project manager has final responsibility for all administrative and budgetary matters. She is responsible for assigning tasks and ensuring that the organization is operating in the most efficient and coordinated manner. The technical manager is accountable for integrating and compiling all technical and policy information, and acts as a liaison between the individual team leaders and the project manager. The individual team leaders are charged with the coordinating and timely reporting of all tasks assigned to the team.

### 6.1 Project Schedule

The project schedule is shown in Figure 42. Several completion dates were postponed for various reasons which included the attendance of 'A Short Course Dealing with the Growing Challenge of Orbital Debris' presented by Southwest Research Institute and difficulty obtaining information on some of the advanced technologies being studied.

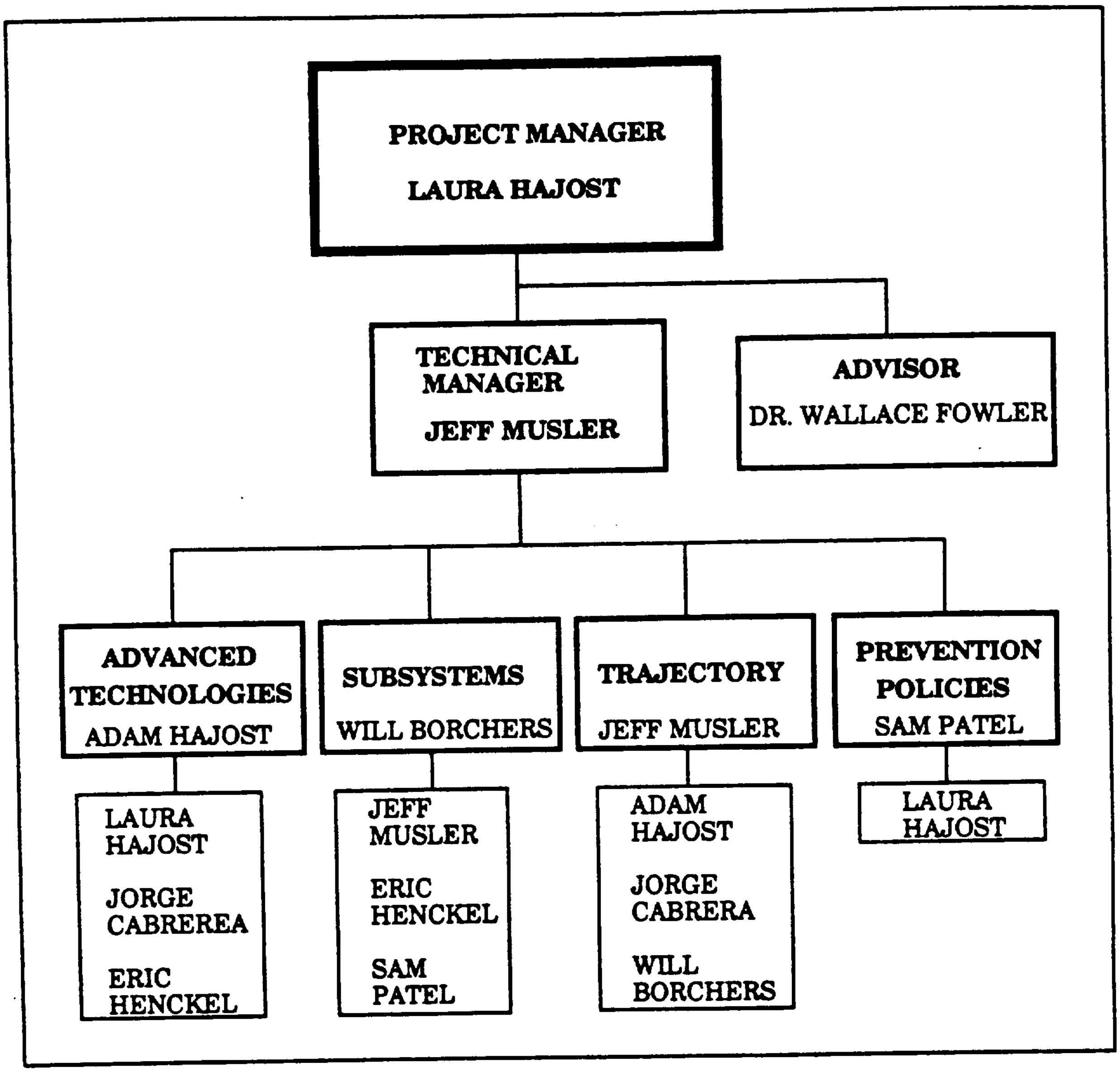


**Figure 41. STRES, Inc Organizational Structure**

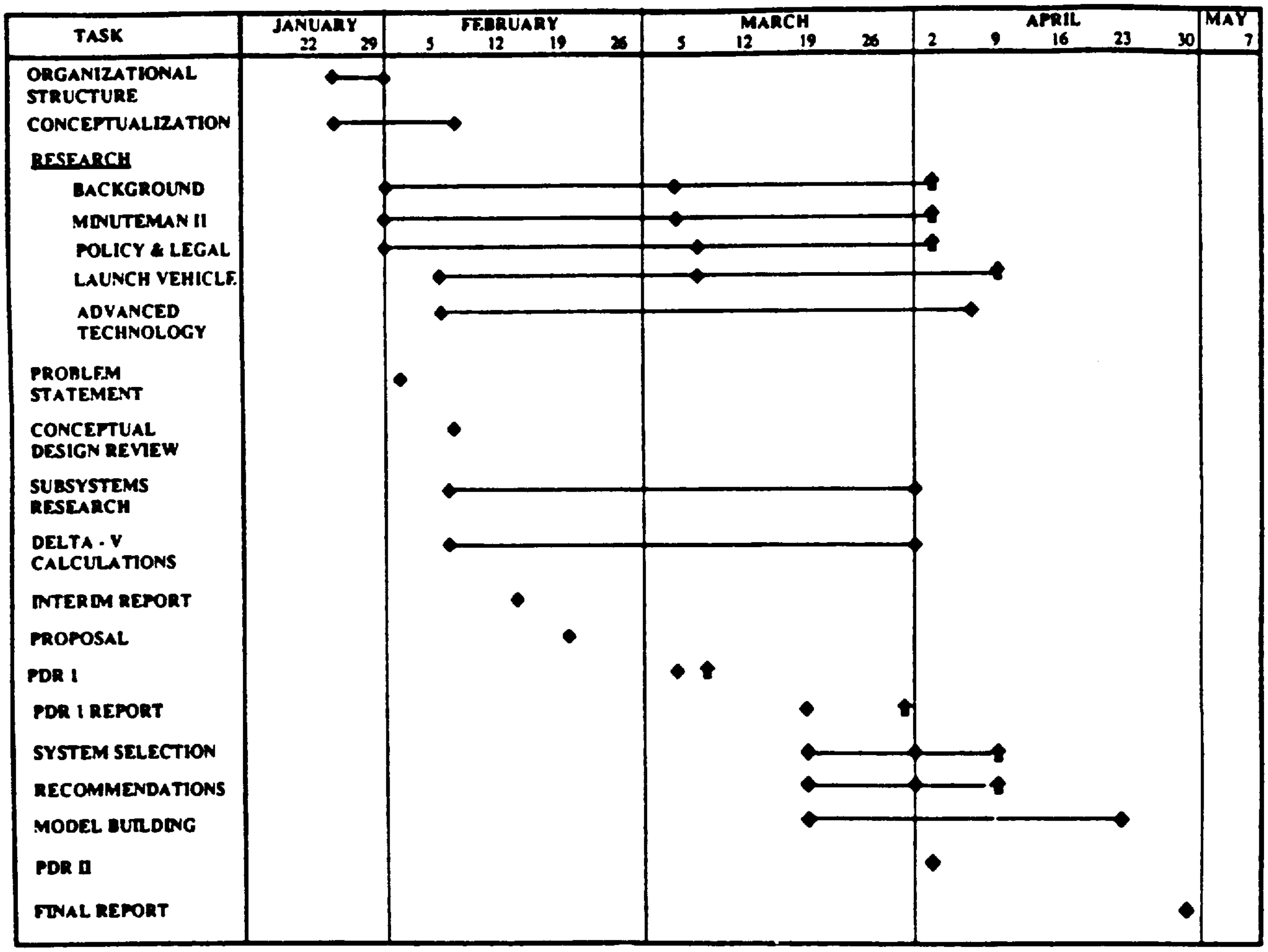


Figure 42. Revised Project Schedule

## 6.2 Cost Report

Table 10 shows the initial projected manpower and materials costs. Comparing these figures with the total expenses shown in Table 11 indicates the project was not completed within the expected budget. The number of man hours required to complete the project was grossly underestimated which resulted in the cost overrun. However, since the contract was awarded on a cost plus basis, the overrun will not pose a problem.

July 24, 1990:

Pages 77 and 78 removed because of funding information.

PHILIP N. FRENCH
Document Evaluator

## 7.0 REFERENCES

### 7.1 References for Introduction

None

### 7.2 References for Orbital Debris Background and Severity


1. Baker, H. A., *Space Debris: Legal and Policy Implications*, Martinus Nijhoff Publishers, 1989.

2. "Contamination Threatens ASAF Payload," *Aviation Week and Space Technology*, May 24, 1982, p. 63.

3. Chobotov, V.A., "Classification of Orbits with Regard to the Collision Hazard in Space,"*Journal of Spacecraft and Rockets*, Vol. 20, 1983, p. 484.

4. Cour-Palais, B., "Shielding Against Debris," *Aerospace America*, June, 1988.

5. "Defense Department Unveils $1.2 Billion ASAT Restructuring Plan," *Aviation Week and Space Technology*, Vol. 19, Mar. 16, 1987.

6. Eichler, P, and Rex, D., "Chain Reaction of Debris Generation by Collisions in Space -- A Final Threat to Spaceflight ?", Paper presented at the 40th Congress of the International Astronautical Federation, October 7-12, 1989.

7. Hechler, M., "Collision Probabilities at Low Earth Orbits," *Advances in Space Research*, Vol. 5, 1985, p. 47.

8. Johnson, N.L., "Artificial Satellite Breakups (Part 1): Soviet Ocean Surveillance Satellites," *Journal of British Interplanetary Society*, Vol. 36, 1983, p. 51.

9. Johnson, N.L., and Nauer, D., "History of On-Orbit Satellite Fragmentations", Prepared by TBE, CS88-LKD-001, 3rd edition, October 4, 1987.

10. Johnson, N.L., *Artificial Space Debris* , Malabar, Orbiting Book Company, 1987.

11. Johnson, N.L., "Preventing Collisions in Orbit," *SPACE*, May-June 1987.

12. Kessler, D.J., and Cour-Palais, B.G., "Collision Frequency of Artificial Satellites: Creation of a Debris Belt," *Journal of Geophysical Research*, June 1978, p. 2637.

13. Kessler, D.J., "Collision Frequency of Artificial Satellites,"*Journal of Geophysical Research*, Vol. 83, June 1, 1987.

14. Kessler, D.J., "Space Debris; More Than Meets the Eye," *Sky & Telescope*, June 1987.

15. Loftus, Joseph P., "Orbital Debris from Upper Stage Breakup," *AIAA Progress in Astronautics and Aeronautics*, Washington D.C., 1989.

16. McKnight, Darren S., "Determining the Cause of a Satellite Fragmentation Case Study of the Kosmos 1275 Breakup," Presented at the 38th Congress of International Astronautical Federation, Brighton, UK, 1987.

17. Oberg, A., "Trashing the Orbital Frontier," *Science Digest*, Oct. 1984.

18. Olmstead, D.A., "Orbital Debris Management: International Cooperation for a Growing Safety Hazard", *Space Safety and Rescue*, 1982-83.

19. "Orbital Space Debris Threatens the Future of Scientific and Commercial Missions ,"*Satellite News*, July 18, 1988, p. 3.

20. "Orbiting Junk Threatens Space Missions" *The New York Times*, August 4, 1987, p. C1.

21. Southwest Research Institute Short Course on Dealing with the Growing Challenge of Orbital Debris, March 19-23, 1990.

22. "Space Becoming Earth's Junkyard," *The Montreal Gazette*, June 12, 1988, p. F4.

23. "Space Debris: More than Meets the Eye," *Aerospace America*, June 1987, p. 187.

24. "The Orbiting Junkyard," *Futurist*, April 1982, p. 77.

25. "Used Ariane Stage Explodes, Creating Space Debris Hazard," *Aviation Week and Space Technology*, Dec. 1, 1986, p. 34.

26. Wolfe, M.G., and Temple, L.P. III, "Department of Defense Policy and the Development of a Global Policy for the Control of Space Debris,"*Journal of the International Astronautics Federation*, Oct. 1987.

### 7.3 References for System Selection


1. Galloway, Eilene, "Nuclear Powered Satellites: The U.S.S.R. Cosmos 954 and the Canadian Claim," *Akron Law Review*, Vol. 12, No. 3, pp. 401-415.

2. Kessler, D.J., and Mueller, A. C., "The Effects of Particles from Solid Rocket Motors Fired in Space," Advances in Space Research, Vol. 5, 1985, pp. 77-86.

3. Kessler, D.J., Reynolds, Robert C., and Anzmeador, Philip D., "Orbital Debris Environment for Spacecraft Designed to Operate in Low Earth Orbit," *National Aeronautics and Space Administration, Technical Memorandum 100 471*, April 1989.

4. "Orbital Space Debris", (Hearing before the Subcommittee on Space Sciences and Applications of the Committee on Science, Space, and Technology House of Representatives: Ninety-ninth Congress, Second Session; Mar. 4, 1986, no. 97) U.S. Government Printing Office, Washington, 1986.

5. Rains, L., "Satellite SRB's Rapidly Depleting Ozone, Soviets Conclude," *Space News*, Feb. 12, 1990, p. 8.

6. Reinhold, R., "Space Junk Emits Clatter From Coast to Coast," *New York Times*, October 14, 1987.

7. "Review of RTG Utilization in Space Missions," (Hearings before the Subcommittee on Energy Research and Production of the Committee on Science, Space, and Technology House of Representatives; Ninety-ninth Congress, Second Session; Mar. 4, 1986, No. 97) U.S. Government Printing Office, Washington, 1986.

8. "Stratospheric Ozone Depletion," (Hearings before the Subcommittee on Natural Resources, Agriculture Research, and Environment of the Committee on Science, Space, and Technology House of Representatives; One Hundredth Congress, First Session; Mar. 10-12, 1987; No. 53) U.S. Government Printing Office, Washington, 1988.


### 7.4 References for Debris Removal Concept


1. Bachman, C.G., *Laser Radar Systems and Techniques*, Artech House, Inc, 1979.

2. Bouquet, F.L., Cotts, D.B., Frederickson, A.R., and Wall, J.A., *AIAA Spacecraft Dielectric Material Properties and Spacecraft Charging*, Progress in Astronautics and Aeronautics, Monmouth Junction, New Jersey, 1986.

3. Buckley, John D., and Stein, Bland A., *Joining Technologies for the 1990's*, Noyes Data Corporation, Park Ridge, New Jersey, 1986.

4. Johnson, N.L., *Artificial Space Debris*. Malabar, Orbiting Book Company, 1987.

5. Kaminow, Ivan P. and Siegman, A., "Laser Devices and Applications," *IEEE Press*, 1973.

6. Kessler, Donald J. and Su, Shin-Yi, "Orbital Debris," *National Aeronautics and Space Administration Conference Publication 2360*, March 1984.

7. Krokhin, O.N., Arrechi, F.T., and Schultz-Dubois, *Laser Handbook*, North Holland, 1972.

8. Lee, Lieng Huang, *Adhesives, Sealants, and coatings for Space and Harsh Environments*, Plenium Press, New York, 1988.

9. Lees, W.A., *Adhesives in Engineering Design*, The Design Council, London, UK, 1984.

10. Loftus, Joseph P. *Orbital Debris from Upper Stage Breakup*, AIAA Progress in Astronautics and Aeronautics, Washington D.C., 1989.

11. Manhart, S. and Dyrna, P., "Laser Radar Technology and Applications," *TISOE*, 1986.

12. Mathews, J.M., *TRW Space Log*, TRW, Inc., 1974.

13. Rosen, Alan, *Spacecraft Charging by Magnetospheric Plasmas*, TRW, Inc., Princeton, New Jersey, 1975.

14. Sherman, Madeline W., *TRW Space Log*, TRW, Inc., 1983.

15. Southwest Research Institute Short Course on Dealing with the Growing Challenge of Orbital Debris, March 19-23, 1990.

16. User's Guide for the Orbital Maneuvering Vehicle, National Aeronautics and Space Administration, Marshall Space Flight Center, Alabama, June 1989.

17. Weast, Robert C., Handbook of Chemistry and Physics, 65th Edition, CRC Press, Inc., Boca Raton, Florida, 1984.

### 7.5 References for Prevention Concepts

1. Convention on the Liability of Damage Caused by Space Objects, *United Nations General Assembly Resolution 2777 (26)*, Nov. 29, 1971.

2. Convention on the Registration of Objects Launched into Outer Space, 28 U.S.T. 695, T.I.A.S. No. 8480, U.N.t.s. 15, October 9, 1973.

3. Cour-Palais, Burton, G. and Crews, Jeanne Lee, "A Multi-Shock Concept for Spacecraft Shielding," *International Journal of Impact Engineering*, Vol. 10, 1990.

4. "Debris-The Pollutant of Outer Space," *Webster University Press Journal*, February 1987.

5. Johnson, N.L., *Artificial Space Debris* , Malabar, Orbiting Book Company, 1987.

6. Kolossov, Y.M., "Legal Aspects of Outer Space Environmental Protection," *Colloquium of the Law of Outer Space*, Vol. 23, 1980, p. 103.

7. Loftus, Joseph P., *Orbital Debris from Upper Stage Breakup*, AIAA Progress in Astronautics and Aeronautics, Washington D.C., 1989.

8. Reis, H., "Some Reflections on the Liability Convention for Outer Space," *Journal of Space Law*, 1978, p. 6.

9. Schonberg, William P., and Taylor, Roy A., "Oblique Hypervelocity Impact Response of Dual-Sheet Structures," *NASA Technical Memorandum 100358*, 1989.

10. Southwest Research Institute Short Course on Dealing with the Growing Challenge of Orbital Debris, March 19-23, 1990.

11. "Standstill on Orbital Debris Policy," *Aerospace America*, June 1988, p. 8.

12. Wiewiorowska, K., "Some Problems of State Responsibility in Outer Space Law," *Journal of Space Law*, vol. 7, 1979-1980, p. 23.

## Appendix A. Method Used to Select Spacecraft Subsystems

The same basic process was used to determine the proper subsystems required for each space debris removal system. Some of the determining factors of a system are size, weight, expected lifetime, current technology, and safety. Size and weight are usually the driving factor for any design. The subsystem must be as light weight and small as possible, but still able to perform the required mission. Another important deciding factor in the selection of a subsystem is its lifetime. Not only does this affect the overall cost of the system but also the effectiveness of the system. A third contributor to the cost of a subsystem is the level of available technology. The initial costs of developing new technology are extremely high, and the time required is prohibitive. The last controlling factor is safety. The potential hazards and risks of a subsystem must be weighed against its benefits before assessing its actual operational costs.

For the purposes of this project, it was decided that there would be six major subsystems that would require further study. These systems include power, propulsion, data processing, thermal, communication, and launch vehicle.

### A.1 Power Systems

There are several power generation systems that were considered for each space debris removal system. Several of these systems were eliminated because they were unrealistic for the required needs of the removal system. For example, it was determined that solar dynamic power would be not before the removal systems would need to be operational. Nuclear systems have a long life, but the re-entry of an active nuclear system has adverse effects on the earth's atmosphere. Solar photovoltaic systems are considered safe and have a long life, but they are subject to degradation due to atomic oxygen, are extremely large, and due to the size are more likely to be damaged by small orbital debris than other power systems. Batteries are fairly inexpensive, but they are comparatively heavy and have a limited lifetime.

From all of the requirements stated in the selection criteria, it was determined that normal lead-acid batteries would be sufficient for the short missions anticipated for the deorbit devices. It was also determined that normal solar photovoltaic power would be sufficient all of the other removal systems under consideration. Batteries would also be need for the solar photovoltaic systems in order to produce power when the satellite is in the shade.

## A.2 Propulsion Systems

The two viable power systems are solid motors and liquid engines, because electric and photon propulsive subsystems are still under development; hence, the technology may not be available before the expected launch date, and subsystem reliability may be unproven.

The solid motors are both heavy and dirty. According to Donald J. Kessler, solid motors have significantly damaged the orbital environment and are responsible for a large portion of penetrations from orbital debris. Another consideration is the current inability of all solid motors to restart. This inadequacy is important if the system will not remain in a single altitude for the duration of its life.

Liquid engines are much cleaner than the solid motors. Another significance is the ability of most engines to restart. Not only does this allow a system to change altitudes, but it will also be practical and cost efficient for the eventual deorbit or disposal of the system upon completion of its useful lifetime.

For these reasons, it was apparent that the best propulsion systems for the debris removal system would be the liquid propellant system. Specifically, the hypergolic propellants monomethylhydrazine and nitrogen tetroxide were chosen because these propellants have a long history of successful use.

## A.3 Data Processing System

A spacecraft computer system has two major functions. First, the computer coordinates communications between the spacecraft and the ground. The communications may be telemetry or scientific data gathered by the spacecraft. By giving instructions to the guidance and propulsion systems the computer makes sure that the course and attitude of the spacecraft are correct. Second, the computer coordinates the activities of the various subsystems.

Using various resources, it was determined that the data processing system needed by the relatively unintelligent systems such as the umbrella satellite and the deorbit devices would be on the order of 5 kg. However, the masses of the data processing system needed by the highly advanced laser beam unit and repulsor satellite are closer to 25 kg.

### A.4 Thermal Control

One unfortunate by-product of every data processing system is the generation of heat. In order to insure the reliability of the data processing system, a thermal control system must be employed to protect the sensitive computers from the excess heat. Therefore, where necessary, the removal system incorporates standard radiators to eliminate heat. This system works by passing a liquid with a high heat capacity, such as Freon, through lines that surround the heat generating devices. This process increases the temperature of the Freon and disapates the heat generated by the avionics. Theses loops then flow into a radiator that disposes the excess heat into space. As a result the Freon is cooled and is returned to the heat generating devices for additional cooling. These types of systems are both highly effective and well proven.

### A.5 Communications

Several of the debris removal systems investigated required teleoperation and highly advanced tracking. For the roving vehicle and the single rendezvous return vehicle, the weights of the tracking and communications systems were already included in the baseline designs for the orbital maneuvering vehicle and the proximity operations vehicle on which these space debris removal systems were based.

In addition it was determined that the capability to transmit to the tracking and data relay satellite (TDRS) should be provided for each system that required teleoperations. The TDRS system is already proven and in place, therefore, no additional development cost will be incurred through this selection. It should also be noted that complete orbital coverage is not provided by the TDRS system. As a result, operations will have to be interrupted or automated during the where there is no TDRS coverage.

For the resupply base system, the mass of the communications system was estimated to be around 60 kg. This value was chosen because it represented a mid-range value for the estimated required mission.

A.6 Launch Vehicle

The main concerns when selecting a launch vehicle is its capabilities and effect on the environment. A launch vehicle must be able to place the system selected into the necessary orbit., but a launch system cannot have a catastrophic effect on the earth and orbital environments. Included in this concern is the detrimental effect on the ozone layer and the contribution to the current orbital debris levels.

The Minuteman II was the first launch vehicle considered because it will soon be decommissioned and will probably be attained cheaper than other launch vehicles. Delta-V calculations for the Minuteman II show that it can only lift 600 kg, which is well below the mass of any systems under consideration, into a 500 km orbit. If the Minuteman II is augmented with 6 Castor Rockets, the payload will increase to 950 kg, which will carry a few of the subsystem into orbit. In addition to the small payload capacities, the Minuteman II emits chloroflourocarbons during launch. A report by Rand Corporation estimated U.S. consumption of chloroflourocarbons (CFCs) and other harmful ozone substances at over 113,000 metric tons per year. If the Minuteman II configuration is used, approximately 6,800 metric tons of ammonium perchlorate would be released annually, provided that a schedule of 40 launches per year could be attained. Therefore, the addition of ozone harming elements from a space debris removal system produces would be minimal. In the case of a Minuteman II and a six castor rocket arrangement, 70% of the solid fuel consists of ammonium perchlorate. This chemical is extremely dangerous to humans, and its by-products are destroyers of ozone. This and other substances have drawn attention amid recent concerns over the growing depletion of the earth's ozone layer. On the basis of the small launch payloads and the emission of harmful elements into the environment, the Minuteman II does not appear to be a premium choice for the launch vehicle.

Because of the Minuteman IIs launch characteristics, several other launch vehicles are under consideration. These vehicles are the Atlas Centaur, Delta 3920, Delta 6920, Titan 4, and Space Shuttle. These launchers have greatly improved payload capacities and are much cleaner than the Minuteman II.

## Appendix B. - Fuel Mass Calculations for the Deorbit Device

The Deorbit Device (DD) is used to deorbit large pieces of space debris. Since the average weight of space debris was computed to be 1000 kg and the most densely populated orbits are between 500 km and 1,000 km, it was decided to find a current technology engine that would be capable of deorbiting at least a 1100 kg satellite at 1000 km to a 100 km perigee orbit, which would accelerate re-entry due to atmospheric drag. Sample calculations were made assuming that the debris was in a circular orbit and that the DD would impart an ideal delta-v. An iterative process was performed to find the engine that best matched our criteria. The engine that emerged from the iterations was the OMS Engine from the Space Shuttle. In addition to the engine, navigation, attitude and reaction control, power, and structure subsystems were added to complete the Deorbit Device.

### Parameters

r1= initial orbit radius of debris and DD upon rendezvous=7378 km(1000 km altitude)
r2= orbit perigee of debris and deorbit device after transfer=6473 km(100 km altitude)
u = earth gravitational parameter = 3.986 E+5 km^3/s^2
Mass Deorbit Device (MDD) = 182 kg
Mass of Fuel on Deorbit Device (MFDD) = 100kg
c = exhaust velocity for OMS engine = 3040 m/s
Mass of Satellite (MS) = ?

### Calculate the Delta-V Necessary for Deorbit

Calculate the Transfer Orbit's Semi-Major Axis
at = semi-major axis of transfer orbit = (r1+r2)/2 = 6928 km

Calculate Circular Velocity at 1000 km
Vc1=sqrt(u/r1)=7.350 km/s

Calculate Perigee Velocity for Transfer
Vpt=sqrt(u*(2/r1 - 1/at))=7.107 km/s

Calculate Delta-V
delta-v = Vc1-Vpt = .243 km/s = 243 m/s

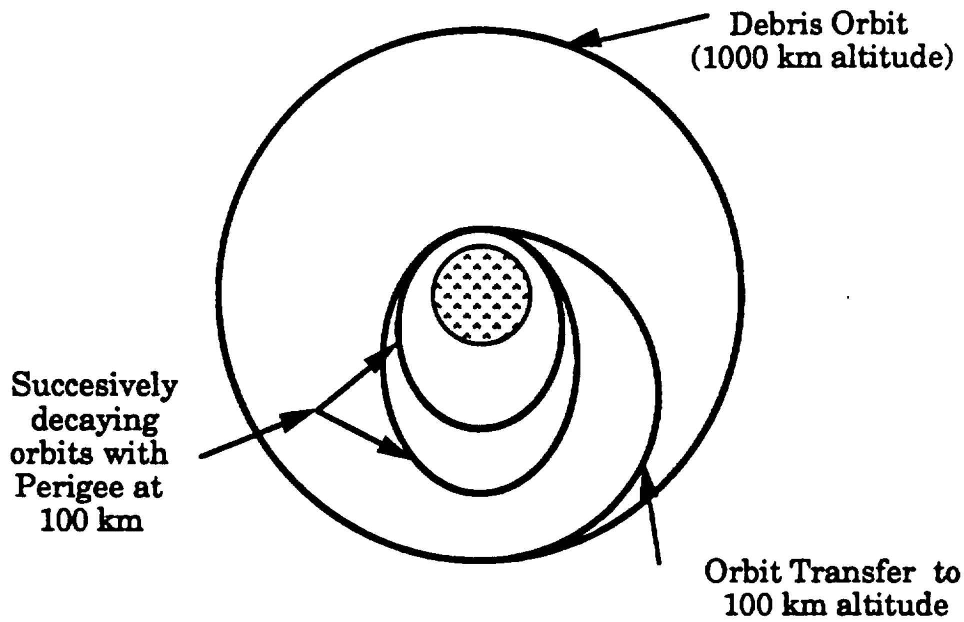


Figure D.1 - Debris and Deorbit Device Transfer Orbit to Obtain 100 km Perigee for Succesively Decaying Orbits

**Calculate the Mass of the Satellite**

Use Ideal Delta-V Equation:

delta-v = c*ln(Initial Mass/Final Mass)

delta-v = c*ln[(MDD+MS)/(MDD-MFDD+MS)]

e^(delta-v/c)=[(MDD+MS)/(MDD-MFDD+MS)]

1.083 = [(182 kg + MS)/(182 kg - 100 kg + MS)]

1.083*(82 kg + MS) = 182 kg + MS

.083 MS = 93.2 kg

<u>MS = 1100 kg</u>.

## APPENDIX C

## Calculations for Proposed Active Removal Systems Sample Mission

Proximity Operations

Cauchy-Wilshire (C-W) equations are used to describe the motion of one spacecraft moving with respect to another. Because of this characteristic, C-W equations are ideal for determining the time-of-flight, position and velocity during proximity operations. The C-W equations are summarized in matrix form below.

$$
\begin{vmatrix} x(t) \\ y(t) \\ z(t) \\ v_x(t) \\ v_y(t) \\ v_z(t) \end{vmatrix}
=
\begin{vmatrix}
1 & 0 & 6\omega t-6\sin(\omega t) & \frac{4\sin(\omega t)-3\omega t}{\omega} & 0 & \frac{2-2\cos(\omega t)}{\omega} \\
0 & \cos(wt) & 0 & 0 & \frac{\sin(\omega t)}{w} & 0 \\
0 & 0 & 4-3\cos(\omega t) & \frac{2\cos(\omega t)-2}{\omega} & 0 & \frac{\sin(\omega t)}{\omega} \\
0 & 0 & 6\omega(1+\cos(\omega t)) & 4\cos(\omega t)-3 & 0 & 2\sin(\omega t) \\
0 & -w\sin(wt) & 0 & 0 & \cos(\omega t) & 0 \\
0 & 0 & 3\omega\sin(\omega t) & -2\sin(\omega t) & 0 & \cos(\omega t)
\end{vmatrix}
\begin{vmatrix} x_o \\ y_o \\ z_o \\ v_{xo} \\ v_{yo} \\ v_{zo} \end{vmatrix}
$$

$$w = ( g_e r_e^2 / r_o^3)^{1/2} \qquad (1)$$

In order to use these equations for targeting, it is necessary to assume that the two spacecraft are in circular and coplanar orbits. In addition, one of the two objects must be the center of a Local Vertical - Local Horizon (LVLH) set of coordinates with the x-axis in the direction of the velocity vector, the y-axis in the direction opposite to the orbital angular momentum vector, and the z-axis pointing radially toward the inertial coordinate center. A further requirement includes having at least 7 initial conditions. Since there are only 7 equations and 14 unknowns, 7 variables (position, velocity, time, and/ or distance from the inertial origin) must be initially given in order to solve the equations without having to perform complicated numerical analyses.

For the sample mission described in the text, the Earth's center serves as the inertial coordinate center and the LVLH origin lies on the orbiting debris satillite. It is necessary to choose the debris as the origin because it is traveling in a constant orbit whereas, the roving vehicle is constantly changing position during the proximity operations. For the initial conditions, we chose the initial and final positions of the roving vehicle ($x_o, y_o, z_o, x_f, y_f, z_f$) and the time-of-flight (t) between those two positions.

After coding in the C-W equations and initial conditions into a TK! model, initial and final rendezvous velocities were solved for and the total $\Delta V$ for the proximity operations determined. In order to determine an optimum $\Delta V$ for rendezvous, time-of-flight was varied from 10 minutes to 25 hours and compared with resulting $\Delta V$s. This procedure was performed for each of the 8 redezvouses during the course of the sample mission. The TK! model and a sample run for one of the proximity operations appears at the end of this Appendix.

Orbit Transfers

Orbit transfers were carried out in the sample mission by using a combination of plane changes and Hohmann transfers. The amount of $\Delta V$ required to make a transfer between two circular and nonplanar orbits is equated below.

$$\Delta V1 = [\, V_{pt}^2 + V_{c1}^2 - 2V_{pt}V_{c1} \cos(\Delta i_L) \,]^{1/2}$$
$$\Delta V2 = [\, V_{at}^2 + V_{c2}^2 - 2V_{at}V_{c2} \cos(\Delta i_H) \,]^{1/2}$$

where

$$V_{pt} = [\, \mu\, (\, 2/r1 - 1/a_t) \,]^{\,1/2}$$
$$V_{c1} = [\mu/r_1]$$
$$V_{at} = [\, \mu(\, 2/r_2 - 1/a_t) \,]^{\,1/2}$$
$$V_{c2} = [\mu/r_2]$$

Because simple plane changes are very expensive, it is necessary to minimize the amount of $\Delta V$ necessary to transfer from one orbit to another. This can be done by varying the amount of plane change ($\Delta i$) performed at each burn. As it turns out, it is more efficient to do most of the required plane change at higher altitudes; however, maximum efficiency does not occur at the highest point in the transfer orbit. Therefore, the above equations were coded into a TK! model and solved with varying magnitudes of inclination change. The resulting $\Delta V$s were then compared with the corresponding change in inclination at the specified point in the transfer orbit, and the minimum $\Delta V$ determined. The TK! model and a sample run for one of the transfer orbits with inclination change appear at the end of this Appendix.

Fuel Analysis for an Eight Rendezvous Mission for a Roving Vehicle

Assumptions:

- All orbit transfers are accomplished via Hohmann transfer,
- All rendezvous procedures require a total DV of 50 m/s,
- All deorbit packages are 182 kg,
- Propellent (Hydrazine) $I_{sp}$ = 309.9 s,
- Roving vehicle without deorbit devices is 9628 kg.

Formulas:

Ideal Rocket:

$$\Delta V = c \ln(M_0/M_f)$$

$$c = gI_{sp}$$

$$\Delta V_1 + \Delta V_2 + \ldots + \Delta V_n = c \ln(M_0/M_1) + c \ln(M_1/M_2) + \ldots + c \ln(M_{n-1}/M_n)$$

$$\sum \Delta V_i = c \ln(M_0/M_n)$$

Mass Analysis:

$M_{TOT}$ before mission: 11,084 kg

500 km parking orbit to 862 km (1.5° plane change)

$\Delta V + \Delta V_R = 293$ m/s

$M_f = 10{,}065.9$ kg — $M_{fuel} = 1018.1$ kg

attach 182 kg package

$M_f = 9883.9$ kg

862 km to 1190 km (0.8° plane change)

$\Delta V + \Delta V_R = 180.2$ m/s

$M_f = 9315.2$ kg — $M_{fuel} = 568.7$ kg

attach 182 kg package

$M_f = 9133.2$ kg

1190 km to 1130 km (0.5° plane change)

$\Delta V + \Delta V_R = 86.3$ m/s

$M_f = 8877.7$ kg — $M_{fuel} = 255.5$ kg

attach 182 kg package

$M_f = 8695.7$ kg

1130 km to 1120 km (0.5° plane change)

$\Delta V + \Delta V_R = 79.6$ m/s

$M_f = 8471.0$ kg — $M_{fuel} = 224.7$ kg

attach 182 kg package

$M_f = 8289.0$ kg

1120 km to 1250 km (0.2° plane change)

$\Delta V + \Delta V_R = 82.5$ m/s

$M_f = 8067.2$ kg — $M_{fuel} = 221.8$ kg

attach 182 kg package

$M_f = 8{,}381.8$ kg

1250 km to 937 km (0.7° plane change)

$\Delta V + \Delta V_R = 194.1$ m/s

$M_f = 7397.6$ kg — $M_{fuel} = 487.6$

attach 182 kg package

$M_f = 7215.6$ kg

937 km to 894 km (1.3° plane change)

$\Delta V + \Delta V_R =$ 186.5 m/s

$M_f = 6786.3$ kg $M_{fuel} = 429.2$ kg

attach 182 kg package

$M_f = 6604.4$ kg

894 km to 635 km (0.5° plane change)

$\Delta V + \Delta V_R =$ 169.6 m/s

$M_f = 6246.2$ kg $M_{fuel} = 358.2$ kg

attach 182 kg package

$M_f = 6064.2$ kg

Mass Analysis Summary:

Roving Vehicle: Final Mass = 6064.2 kg

Total Amount of Fuel Used = 3564.0 kg

Total Mass of 8 Deorbit Devices = 1,456.0 kg

Total $\Delta V$ Imparted = 1271.8 m/s

Percentage Fuel Used = 87.3%

TK! Model

**and**

Sample Run for Rendezvous with Satellite in 1250 km orbit

RULE SHEET

```
S Rule
  XF = A11*XOF + A13*ZOF  + B11*XDOF + B13*ZDOF
  YF = A22*YOF + B22*YDOF
  ZF = A33*ZOF + B31*XDOF + B33*ZDOF

  XDF= C13*ZOF + D11*XDOF + D13*ZDOF
  YDF= C22*YOF + D22*YDOF
  ZDF= C33*ZOF + D31*XDOF + D33*ZDOF

  DELTAVF  = SQRT(XDF^2 + YDF^2 + ZDF^2)
  DELTAVO  = SQRT(XDOF^2 +YDOF^2 +ZDOF^2)

  W    = SQRT(G*RE^2/RO^3)

  A11  = 1
  A13  = (-6*SIN(W*T)+6*W*T)
  A22  = COS(W*T)
  A33  = (-3*COS(W*T)+4)

  B11  = (4/W*SIN(W*T)-3*T)
  B13  = (-2/W*COS(W*T)+2/W)
  B22  = SIN(W*T)/W
  B31  = (2/W*COS(W*T)-2/W)
  B33  = SIN(W*T)/W

  C13  = (-6*W*COS(W*T)+6*W)
  C22  = -W*SIN(W*T)
  C33  = 3*W*SIN(W*T)

  D11  = (4*COS(W*T)-3)
  D13  = 2*SIN(W*T)
  D22  = COS(W*T)
  D31  = -2*SIN(W*T)
  D33  = COS(W*T)
  XF = A11*XOF + A13*ZOF  + B11*XDOF + B13*ZDOF
  YF = A22*YOF + B22*YDOF
  ZF = A33*ZOF + B31*XDOF + B33*ZDOF
  XDF= C13*ZOF + D11*XDOF + D13*ZDOF
  YDF= C22*YOF + D22*YDOF
  ZDF= C33*ZOF + D31*XDOF + D33*ZDOF

  DELTAVF  = SQRT(XDF^2 + YDF^2 + ZDF^2)
  DELTAVO  = SQRT(XDOF^2 +YDOF^2 +ZDOF^2)
  W    = SQRT(G*RE^2/RO^3)
  A11  = 1
  A13  = (-6*SIN(W*T)+6*W*T)
  A22  = COS(W*T)
  A33  = (-3*COS(W*T)+4)
  B11  = (4/W*SIN(W*T)-3*T)
  B13  = (-2/W*COS(W*T)+2/W)
  B22  = SIN(W*T)/W
  B31  = (2/W*COS(W*T)-2/W)
  B33  = SIN(W*T)/W
  C13  = (-6*W*COS(W*T)+6*W)
  C22  = -W*SIN(W*T)
  C33  = 3*W*SIN(W*T)
  D11  = (4*COS(W*T)-3)
  D13  = 2*SIN(W*T)
  D22  = COS(W*T)
  D31  = -2*SIN(W*T)
  D33  = COS(W*T)
  DVTOT=DELTAVO+DELTAVF
```

| St | Input | Name | Output | Unit | Comment |
|---|---|---|---|---|---|
| L | 0 | XF | | | X-POSITION |
| L | 0 | YF | | | Y-POSITION |
| L | 0 | ZF | | | Z-POSITION |
| | 10000 | XOF | | | INITIAL X-POSITION |
| | 0 | YOF | | | INITIAL Y-POSITION |
| | 10000 | ZOF | | | INITIAL Z-POSITION |
| L | | XDF | -6.7643 | | X-VELOCITY |
| L | | YDF | 0 | | Y-VLEOCITY |
| L | | ZDF | 8.5427931 | | Z-VELOCITY |
| L | | XDOF | 12.695041 | | INITIAL X-VELOCITY |
| L | | YDOF | 0 | | INITIAL Y-VELOCITY |
| L | | ZDOF | -15.54254 | | INITIAL Z-VELOCITY |
| L | | DELTAVF | 10.896563 | | DELTAV TO STOP |
| L | | DELTAVO | 20.068247 | | INITIAL DELTAV |
| L | 1946.9388 | T | | | TIME |
| | 9.81 | G | | | GRAVITATIONAL ACCELERATION |
| | 6378145 | RE | | | RADIUS OF THE EARTH |
| | 7628145 | RO | | | INITIAL RADIUS OF OBJECT |
| | | W | .00097297 | | |
| | | A11 | 1 | | |
| | | A13 | 5.6770931 | | |
| | | B11 | -1942.945 | | |
| | | B13 | 2709.0277 | | |
| | | A22 | -.3178974 | | |
| | | B22 | 974.46786 | | |
| | | A33 | 4.9536921 | | |
| | | B31 | -2709.028 | | |
| | | B33 | 974.46786 | | |
| | | C13 | .00769362 | | |
| | | D11 | -4.271589 | | |
| | | D13 | 1.8962503 | | |
| | | C22 | -.0009225 | | |
| | | D22 | -.3178974 | | |
| | | C33 | .00276748 | | |
| | | D31 | -1.89625 | | |
| | | D33 | -.3178974 | | |
| L | | DVTOT | 30.96481 | | TOTAL DELTA-V NEEDED |

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Title: TOTAL DELTA-V AS FN. OF TIME OF FLIGHT (XO=ZO=10KM,RO=7628.145KM)

| Element | TOF(S) | DVINI(M/S) | DVSTP(M/S) | DVTOT(M/S) |
|---|---|---|---|---|
| 1 | 600 | 7.61778164 | 7.61778164 | 15.2355633 |
| 2 | 2351.02041 | 16.907997 | 16.5324873 | 33.4404843 |
| 3 | 4102.04082 | 17.8213568 | 17.0227054 | 34.8440622 |
| 4 | 5853.06122 | 18.1087974 | 16.8078129 | 34.9166103 |
| 5 | 7604.08163 | 20.0190692 | 18.3263838 | 38.345453 |
| 6 | 9355.10204 | 505.367701 | 505.282822 | 1010.65052 |
| 7 | 11106.1224 | 19.4582799 | 16.4647192 | 35.9229991 |
| 8 | 12857.1429 | 19.3729362 | 15.6739844 | 35.0469206 |
| 9 | 14608.1633 | 19.8023797 | 15.5108791 | 35.3132588 |
| 10 | 16359.1837 | 97.152921 | 96.2597636 | 193.412685 |
| 11 | 18110.2041 | 18.7605576 | 12.6178235 | 31.378381 |
| 12 | 19861.2245 | 296.177571 | 295.820981 | 591.998552 |
| 13 | 21612.2449 | 19.893474 | 12.9746223 | 32.8680963 |
| 14 | 23363.2653 | 33.4128136 | 29.5816118 | 62.9944255 |
| 15 | 25114.2857 | 18.5485457 | 9.56028895 | 28.1088346 |
| 16 | 26865.3061 | 28.7344506 | 23.759637 | 52.4940876 |
| 17 | 28616.3265 | 20.2079898 | 11.9001804 | 32.1081702 |
| 18 | 30367.3469 | 21.5024755 | 13.8893187 | 35.3917943 |
| 19 | 32118.3673 | 20.0962581 | 11.5995468 | 31.6958049 |
| 20 | 33869.3878 | 22.8902951 | 16.0454932 | 38.9357883 |
| 21 | 35620.4082 | 21.056827 | 13.4951742 | 34.5520012 |
| 22 | 37371.4286 | 18.8982883 | 10.1610295 | 29.0593178 |
| 23 | 39122.449 | 26.6989193 | 21.6351659 | 48.3340852 |
| 24 | 40873.4694 | 21.1005642 | 14.5258429 | 35.6264071 |
| 25 | 42624.4898 | 24.9776727 | 20.0275575 | 45.0052303 |
| 26 | 44375.5102 | 18.8747781 | 12.0585026 | 30.9332807 |
| 27 | 46126.5306 | 60.9580053 | 59.3041174 | 120.262123 |
| 28 | 47877.551 | 20.1360707 | 14.767492 | 34.9035627 |
| 29 | 49628.5714 | 79.5776453 | 78.4594028 | 158.037048 |
| 30 | 51379.5918 | 19.8777978 | 15.0934925 | 34.9712902 |
| 31 | 53130.6122 | 138.583472 | 138.006837 | 276.590308 |
| 32 | 54881.6327 | 19.4765316 | 15.0264944 | 34.503026 |
| 33 | 56632.6531 | 19.8695051 | 15.6565427 | 35.5260479 |
| 34 | 58383.6735 | 21.7498161 | 18.0287172 | 39.7785333 |
| 35 | 60134.6939 | 37.7245869 | 35.7000878 | 73.4246747 |
| 36 | 61885.7143 | 19.0698425 | 14.564556 | 33.6343985 |
| 37 | 63636.7347 | 18.9509038 | 14.2325739 | 33.1834777 |
| 38 | 65387.7551 | 25.2709039 | 21.7953194 | 47.0662233 |
| 39 | 67138.7755 | 25.0859425 | 21.377251 | 46.4631935 |
| 40 | 68889.7959 | 19.6443256 | 14.2631945 | 33.9075201 |
| 41 | 70640.8163 | 18.9111823 | 12.8058427 | 31.7170251 |
| 42 | 72391.8367 | 34.3918026 | 31.2624824 | 65.6542849 |
| 43 | 74142.8571 | 20.7363968 | 14.5790468 | 35.3154436 |
| 44 | 75893.8776 | 32.3089453 | 28.5428381 | 60.8517833 |
| 45 | 77644.898 | 18.9761225 | 10.9510345 | 29.927157 |
| 46 | 79395.9184 | 106.224723 | 105.041459 | 211.266181 |
| 47 | 81146.9388 | 19.1732688 | 10.4601007 | 29.6333695 |
| 48 | 82897.9592 | 65.285563 | 63.2278936 | 128.513457 |
| 49 | 84648.9796 | 19.0668665 | 9.75587234 | 28.8227388 |
| 50 | 86400 | 42.6784335 | 39.3919161 | 82.0703497 |

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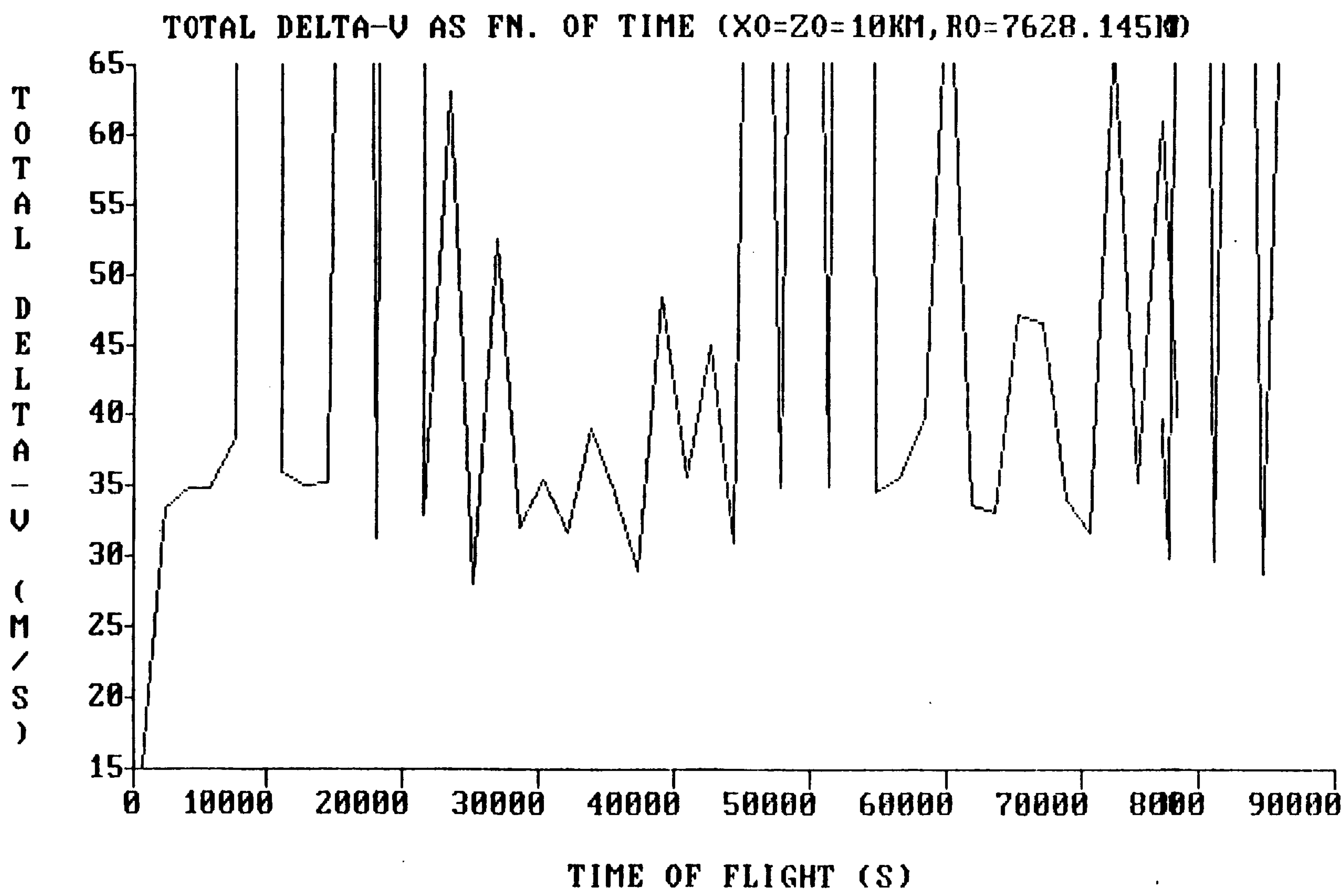
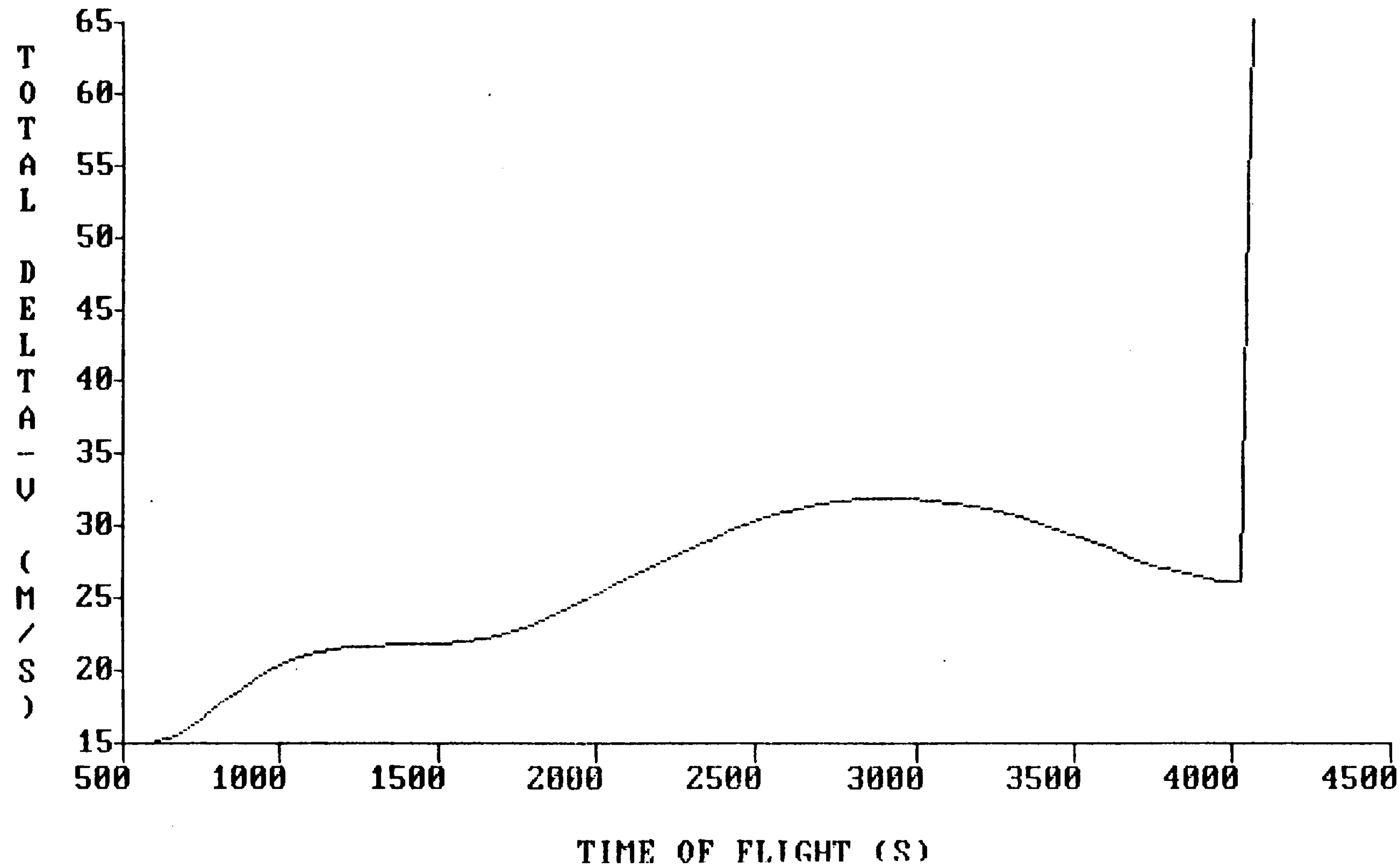

TOTAL DELTA-V AS FN. OF TIME (XO=ZO=10KM,RO=7628.145KM)
TOTAL DELTA-V (M/S)
65
60
55
50
45
40
35
30
25
20
15
0
10000
20000
30000
40000
50000
60000
70000
80000
90000
TIME OF FLIGHT (S)

TABLE: TABLE3

Title: TOTAL DELTA-V AS FN. OF TIME OF FLIGHT (XO=ZO=10KM,RO=7628.145KM)

| Element | TOF(S) | DVINI(M/S) | DVSTP(M/S) | DVTOT(M/S) | XODOT(M/S) | ZODOT(M/S) |
|---|---|---|---|---|---|---|
| 1 | 600 | 7.61778164 | 7.61778164 | 15.2355633 | 1.42678729 | -7.4829723 |
| 2 | 670 | 8.14232649 | 7.33077459 | 15.4731011 | 4.07596556 | -7.0486868 |
| 3 | 740 | 9.09677473 | 7.41100886 | 16.5077836 | 6.22866687 | -6.6298582 |
| 4 | 810 | 10.1427264 | 7.57950724 | 17.7222336 | 8.00729873 | -6.2255976 |
| 5 | 880 | 11.1458623 | 7.70217247 | 18.8480348 | 9.49576854 | -5.8361483 |
| 6 | 950 | 12.0614877 | 7.72633479 | 19.7878225 | 10.7537162 | -5.4623323 |
| 7 | 1020 | 12.8798624 | 7.63892492 | 20.5187873 | 11.8248699 | -5.1052234 |
| 8 | 1090 | 13.6043055 | 7.44582172 | 21.0501273 | 12.7421685 | -4.7659491 |
| 9 | 1160 | 14.2425953 | 7.16272635 | 21.4053216 | 13.5310172 | -4.4455699 |
| 10 | 1230 | 14.8035682 | 6.81158291 | 21.6151511 | 14.2114229 | -4.1450078 |
| 11 | 1300 | 15.2957987 | 6.41968896 | 21.7154876 | 14.7994317 | -3.865007 |
| 12 | 1370 | 15.727131 | 6.02009422 | 21.7472252 | 15.3081213 | -3.6061162 |
| 13 | 1440 | 16.1045643 | 5.65221266 | 21.756777 | 15.7483 | -3.3686851 |
| 14 | 1510 | 16.4342793 | 5.36108032 | 21.7953596 | 16.1290094 | -3.1528705 |
| 15 | 1580 | 16.7217168 | 5.19291747 | 21.9146343 | 16.4578921 | -2.9586487 |
| 16 | 1650 | 16.9716687 | 5.18548349 | 22.1571522 | 16.7414659 | -2.7858317 |
| 17 | 1720 | 17.1883649 | 5.35642904 | 22.5447939 | 16.9853314 | -2.6340853 |
| 18 | 1790 | 17.3755515 | 5.69793939 | 23.0734909 | 17.1943318 | -2.5029477 |
| 19 | 1860 | 17.5365578 | 6.18229393 | 23.7188517 | 17.3726774 | -2.3918488 |
| 20 | 1930 | 17.6743526 | 6.77310733 | 24.4474599 | 17.5240449 | -2.3001278 |
| 21 | 2000 | 17.7915919 | 7.43441369 | 25.2260056 | 17.6516569 | -2.2270499 |
| 22 | 2070 | 17.890659 | 8.13491839 | 26.0255773 | 17.7583465 | -2.1718215 |
| 23 | 2140 | 17.9736968 | 8.84883542 | 26.8225322 | 17.8466106 | -2.1336044 |
| 24 | 2210 | 18.0426365 | 9.55526205 | 27.5978986 | 17.9186546 | -2.1115276 |
| 25 | 2280 | 18.0992205 | 10.2372218 | 28.3364423 | 17.9764299 | -2.1046979 |
| 26 | 2350 | 18.1450224 | 10.8808122 | 29.0258346 | 18.021665 | -2.1122092 |
| 27 | 2420 | 18.1814635 | 11.4745604 | 29.6560239 | 18.0558934 | -2.1331499 |
| 28 | 2490 | 18.2098278 | 12.0089677 | 30.2187955 | 18.0804765 | -2.1666094 |
| 29 | 2560 | 18.2312737 | 12.4762012 | 30.707475 | 18.0966239 | -2.2116835 |
| 30 | 2630 | 18.2468454 | 12.8698888 | 31.1167342 | 18.1054111 | -2.2674781 |
| 31 | 2700 | 18.2574818 | 13.1849881 | 31.4424699 | 18.1077946 | -2.333113 |
| 32 | 2770 | 18.2640248 | 13.4177066 | 31.6817314 | 18.1046255 | -2.4077241 |
| 33 | 2840 | 18.2672268 | 13.5654588 | 31.8326857 | 18.0966616 | -2.4904647 |
| 34 | 2910 | 18.2677573 | 13.6268522 | 31.8946095 | 18.0845773 | -2.5805074 |
| 35 | 2980 | 18.2662081 | 13.6016973 | 31.8679054 | 18.0689732 | -2.6770441 |
| 36 | 3050 | 18.2630992 | 13.4910411 | 31.7541403 | 18.050384 | -2.779286 |
| 37 | 3120 | 18.2588833 | 13.2972261 | 31.5561093 | 18.0292857 | -2.8864639 |
| 38 | 3190 | 18.25395 | 13.0239801 | 31.2779301 | 18.0061025 | -2.9978269 |
| 39 | 3260 | 18.2486302 | 12.6765474 | 30.9251776 | 17.9812115 | -3.1126417 |
| 40 | 3330 | 18.2431993 | 12.2618722 | 30.5050715 | 17.9549487 | -3.2301916 |
| 41 | 3400 | 18.2378808 | 11.7888543 | 30.0267351 | 17.9276129 | -3.3497746 |
| 42 | 3470 | 18.2328495 | 11.2686965 | 29.501546 | 17.8994701 | -3.4707019 |
| 43 | 3540 | 18.2282344 | 10.715362 | 28.9435963 | 17.8707565 | -3.5922961 |
| 44 | 3610 | 18.2241212 | 10.1461448 | 28.370266 | 17.8416822 | -3.7138887 |
| 45 | 3680 | 18.2205552 | 9.58230457 | 27.8028597 | 17.8124338 | -3.8348184 |
| 46 | 3750 | 18.2175431 | 9.049599 | 27.2671421 | 17.7831769 | -3.9544274 |
| 47 | 3820 | 18.2150559 | 8.57832928 | 26.7933852 | 17.7540584 | -4.0720597 |
| 48 | 3890 | 18.2130301 | 8.20221062 | 26.4152407 | 17.7252085 | -4.1870572 |
| 49 | 3960 | 18.2113701 | 7.95522764 | 26.1665977 | 17.6967425 | -4.2987562 |
| 50 | 4030 | 18.2099495 | 7.8661699 | 26.0761194 | 17.6687623 | -4.4064837 |
| 51 | 4100 | 0 | 100.544072 | 100.544072 | 0 | 0 |

TOTAL DELTA-V AS FN. OF TIME (X0=Z0=10KM,R0=7628.145KM)



TK! Model

and

Sample Run for Transfer Orbit from 1120 km to 1250 km

with 0.2° Plane Change

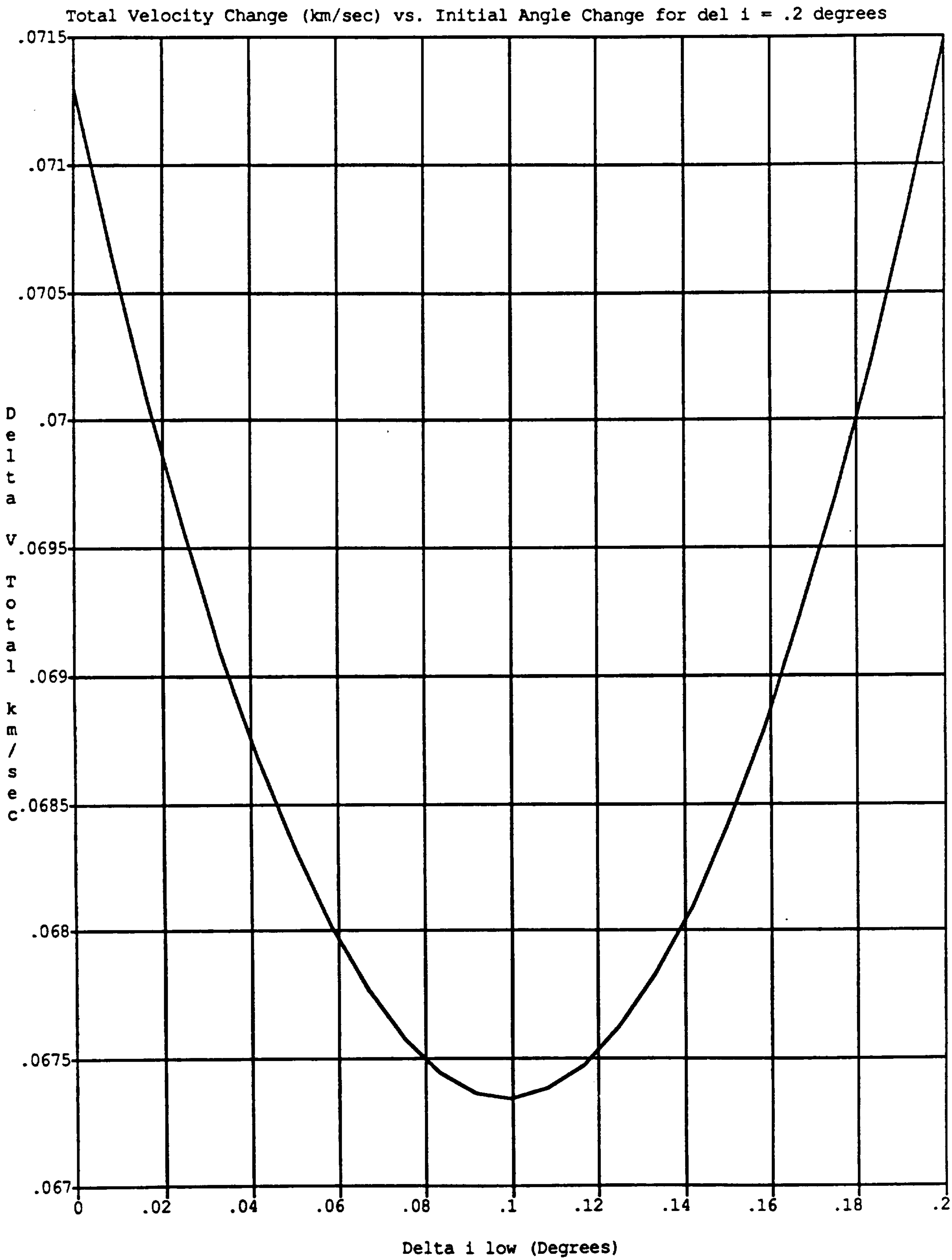
S Rule

```
"This program computes the delta V's for out of plane orbit changes between
"circular orbits.  An iteratbie process is used to find the minimum delta V.
* Vc1 = sqrt(u/r1)
* Vpt = sqrt(u*(2/r1-1/At))
* At = (r1+r2)/2
* delVlo = sqrt(Vpt*Vpt + Vc1*Vc1 - 2*Vpt*Vc1*cos(delilo))
"this completes the equations for the first burn
* Vc2 = sqrt(u/r2)
* Vat = sqrt(u*(2/r2-1/At))
* delihi = i - delilo
* delVhi = sqrt(Vc2*Vc2 + Vat*Vat - 2*Vat*Vc2*cos(delihi))
* delVtot = delVlo + delVhi
```

| St | Input | Name | Output | Unit | Comment |
|---|---|---|---|---|---|
| | | Vc1 | 7.2911541 | km/s | Initial Circular Speed |
| | 398600.5 | u | | km^3/s^2 | Earth's Gravitational Parameter |
| | 7498 | r1 | | km | Initial Orbit Radius |
| | | Vat | 7.2838687 | km/s | Transfer Apogee Velocity |
| | | At | 7503 | km | Transfer Semi-Major Axis |
| | 7628 | r2 | | km | Final Orbit Radius |
| | | delVlo | .00242901 | km/s | Delta V at first burn |
| | | Vpt | 7.2935831 | km/s | Transfer Perigee Velocity |
| L | 0 | delilo | | deg | Plane Change for first burn |
| | | Vc2 | 7.2862969 | km/s | Final Circular Speed |
| | | delihi | 0 | deg | Plane Change for Second Burn |
| | .2 | i | | deg | Inclination Starting from |
| | | delVhi | .0024282 | km/s | Delta V at final burn |
| L | | delVtot | .00485721 | km/s | Total Delta V |

Title: Initial Orbit Inclination Change and Total Velocity for Non-Coplana

| Element | Initial Orbit Inclination Change | Total Delta V for Transfer(km/s) |
|---|---|---|
| 1 | 0 | .0713 |
| 2 | .00833 | .07067 |
| 3 | .01667 | .07009 |
| 4 | .025 | .06956 |
| 5 | .03333 | .06909 |
| 6 | .04167 | .06868 |
| 7 | .05 | .06832 |
| 8 | .05833 | .06802 |
| 9 | .06667 | .06777 |
| 10 | .075 | .06758 |
| 11 | .08333 | .06744 |
| 12 | .09167 | .06737 |
| 13 | .1 | .06735 |
| 14 | .10833 | .06738 |
| 15 | .11667 | .06748 |
| 16 | .125 | .06763 |
| 17 | .13333 | .06783 |
| 18 | .14167 | .06809 |
| 19 | .15 | .06841 |
| 20 | .15833 | .06879 |
| 21 | .16667 | .06922 |
| 22 | .175 | .0697 |
| 23 | .18333 | .07024 |
| 24 | .19167 | .07083 |
| 25 | .2 | .07148 |


Total Velocity Change (km/sec) vs. Initial Angle Change for del i = .2 degrees
Delta V Total km/sec
.0715
.071
.0705
.07
.0695
.069
.0685
.068
.0675
.067
0
.02
.04
.06
.08
.1
.12
.14
.16
.18
.2
Delta i low (Degrees)